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Final Report On

FEASIBILITY STUDY FOR A MULTIPLE SATELLITE SYSTEM

Contract NAS 2-3925

Prepared For

NASA
AMES RESEARCH CENTER
Moffett Field, California

Report SGC 1089R-3

VOLUME I - SUMMARY

SPACE-GENERAL CORPORATION
El Monte, California

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ABSTRACT

The feasibility of deploying multiple satellites in a non-coplanar array for the purpose of defining spatial and temporal variations of the solar wind in the transition region and near-interplanetary space has been investigated. The four-month analysis and design study has found the multiple satellite mission to be technically feasible within the current technology.

The objective of this report is to present the results of the Multiple Satellite Study and summarize the work leading to the conclusions. Detailed analyses may be found in the Appendices to this Summary volume.

The scope of work covered (1) the establishment of science/system interactions and requirements; (2) the development of major mission options and their comparison leading to the selection of the preferred option; (3) detailed analysis of orbit and control alternatives, including analysis and selection of the common orbit, satellite deployment and system control considerations; (4) the investigation of communications and data handling alternatives including ground station considerations; (5) the specification of system integration criteria, e.g., magnetics, reliability and environmental criteria; and (6) a feasibility design study for the preferred concept, including specification of subsystem characteristics and preliminary weights.

The concept selected for design involves a "pallet" (or bus) carrying four satellites launched into a near ecliptic orbit with the nominal apogee at 20 earth radii. The pallet, spinning after separation from the spinning third stage, reorients the spacecraft using an on-board control system. The satellites are then separated from the pallet at appropriate times in the orbit by spin-off and the use of small solid rocket motors, achieving a non-coplanar array. The satellites are commanded to acquire, record and transmit data according to pre-established operating modes. All elements of the operation and design of the multiple satellite system were investigated and found to be feasible within the stated reliability goal.

Since the mission objectives and design concepts for the Multiple Satellite Program have been proven technically feasible, initiation of a Program Definition Phase is recommended. Certain areas should also receive early technical emphasis for further optimization and definition. These areas include (1) an investigation of an integrated scientific instrument complement; (2) optimization of the separation and deployment maneuvers; (3) a detailed analysis and computer simulation of the multiple satellite array; (4) optimization of the reorientation and aspect sensing systems; (5) a design and demonstration of an improved magnetometer boom; and (6) an analysis, design and demonstration of the spin-separation mechanism.

CONTENTS

ILLUSTRATIONS

TABLES

FOREWORD

This final report documents all technical work completed by the Space-General Corporation on the "Feasibility Study for a Multiple Satellite System". It is submitted in partial fulfillment of the requirements of Contract NAS 2-3925. The document consists of two volumes: VOLUME I - SUMMARY, and VOLUME II - APPENDICES.

The following personnel were responsible for major study tasks, and were primary contributors to the preparation of this final report:

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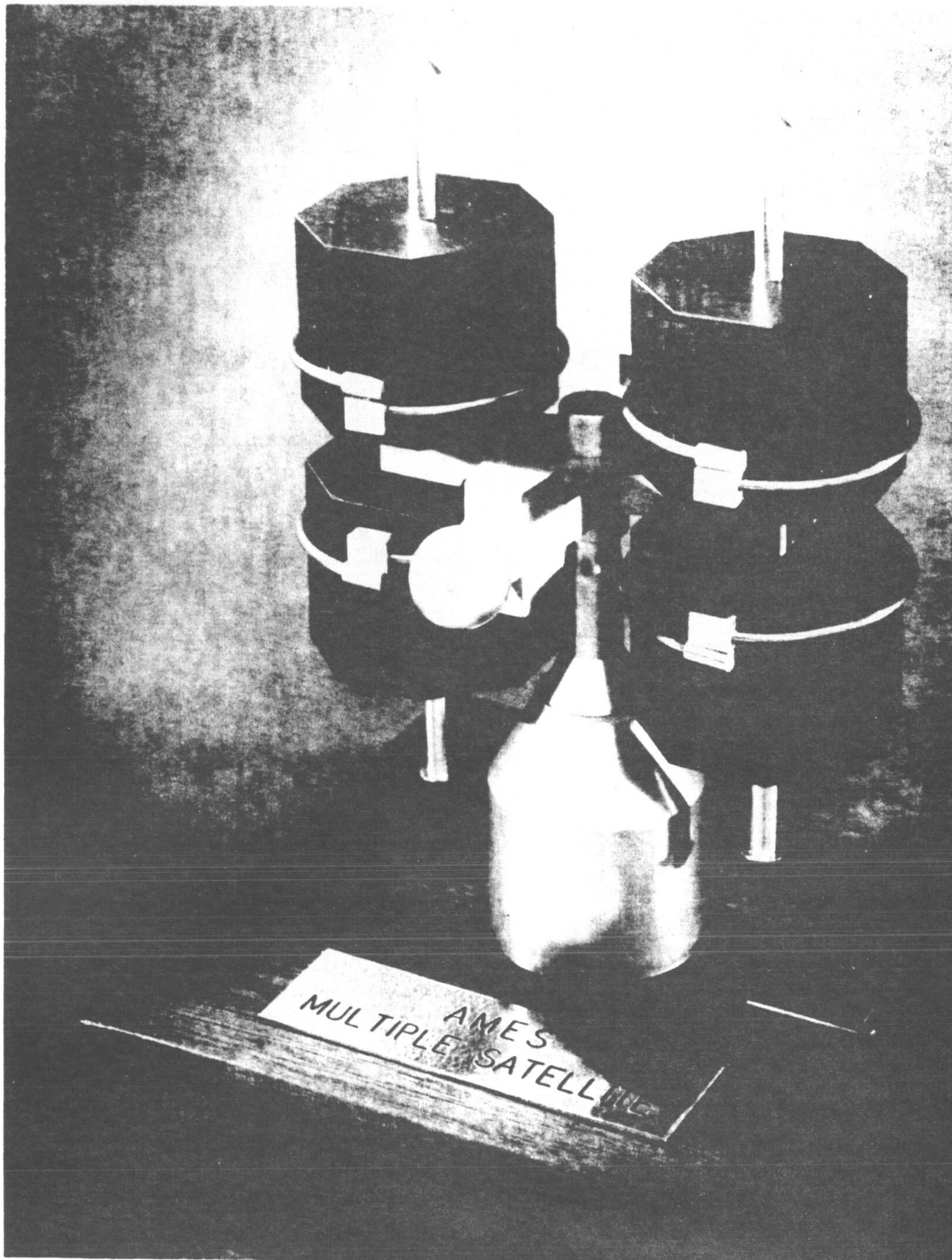

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Section 1

INTRODUCTION

The investigation of the feasibility of using multiple satellites for establishing spatial and temporal variations of the solar wind in the transition region and near-interplanetary space is summarized in this volume. Detailed analyses of all phases of the work are provided in the Appendices.

The ultimate objective of the Multiple Satellite Program is to place four spacecraft in a non-coplanar array, having a highly eccentric nominal orbit, about the earth to traverse the areas of interest -- the magnetosphere, the transition region, and near-interplanetary space. The four spacecraft will acquire magnetic and plasma data in the subsolar region which will allow the separation of time-dependent events from the motion associated with the disturbances being propagated within the plasma. To investigate the feasibility of multiple satellites making the measurements described, five major areas of study have been defined:

a. Science/System Interactions

b. Orbit and Control Considerations

c. Communications and Data Handling

d. System Integration

e. System Design

1.1 SCIENCE/SYSTEM INTERACTIONS

Science/system requirements have been established based on the scientific objectives specified by Ames Research Center. Basic mission requirements were evolved and alternate mission modes established for accomplishing the mission requirements. These major mission option alternatives are evaluated based on the implications of scientific instrument integration, costs, weight, and reliability. Based upon this evaluation, a mission mode utilizing a "pallet" for reorientation of the satellites and their deployment is selected.

1.2 ORBIT AND CONTROL CONSIDERATIONS

Orbit and control considerations are analyzed in depth due to the strong dependence of concept feasibility on the capability of the multiple satellite system to meet the required array end conditions. The investigations include analysis and selection of the common orbit; that is, the nominal orbit from which the satellites are deployed. Launch errors, perturbation effects, and satellite separation effects are considered in the selection of the nominal apogee and perigee altitudes. Launch vehicle capabilities and the available launch window have been established. Analysis was performed to establish the best initial orbit to maximize the coverage of the subsolar line leading to the specification of the orbit inclination and initial lead angle (the initial angle between the major axis and the subsolar line). Deployment sensitivities as a function of position in orbit are determined. These sensitivities are used to specify the separation points in orbit, the separation distances expected, the separation accuracy required and the growth in the satellite array. A computer simulation of the array growth history was conducted to verify the analytical work. The tracking accuracy of the satellites in the array is established. Pallet and satellite control requirements are established, alternate control possibilities are evaluated, and the preferred method is specified. The drift of the satellite spin axis has been predicted through the use of a computer program.

1.3 COMMUNICATIONS AND DATA HANDLING

Communications and data handling for the Multiple Satellite Program presents some unique problems, particularly with respect to long-term data storage, and the interface and availability of the ground stations. The communications and data handling requirements are specified based on a best estimate of the instrument complement. The STADAN network has been specified as the ground station net for the program; thus its availability, capability, and interface characteristics are established. Ground station availability as a function of orbital coverage was determined, through the use of a computer program. Data acquisition duty cycles are estimated and the satellite downlink capability is specified. The required data processing and tape recorder capabilities are determined, and the command functions are specified on a preliminary basis. Alternate antenna design concepts were considered and the preferred designs are selected.

1.4 SYSTEM INTEGRATION

System integration criteria were established including system requirements, limitations, and design goals related to:

a. Magnetics

b. Reliability

c. EMI

d. Environments

In addition, system integration analyses were conducted throughout the program to assure compatibility between scientific requirements and design concepts; between analysis work and the feasibility design study; and between the components of the reference design configuration. An evaluation approach is developed for subsequent use in evaluating specific component and subsystem alternatives. Support systems are considered, emphasizing the support requirements peculiar to the Multiple Satellite Program. A qualification and test philosophy is specified and operational support plan considerations are presented.

1.5 SYSTEM DESIGN

A feasibility design study has been performed for the selected concept. Configuration studies were conducted for the satellites, satellite-pair combinations, the pallet, and the total payload. Design requirements are specified, and the resulting design characteristics including structure and balance are delineated. Specific design analyses include specification of the thermal control design and determination of its capability; specification of power requirements and the recommended power system design, including the effect of boom shadowing on the solar array; comparison of alternative boom designs; description of alternate separation mechanisms; comparison of alternate aspect sensors; and recommendations for the antenna designs. Preliminary weight statements for the total system and the satellites are developed. Initial component surveys were conducted and component availability is specified.

The work conducted in the five basic study areas listed above is summarized in the following sections. Conclusions and recommendations are also defined. The basic conclusion is that the mission objectives and design concepts for the Multiple Satellite Program are technically feasible. As a result, it is recommended that the program should be continued with the implementation of Phase B, the Program Definition Phase.

Section 2

SCIENCE/SYSTEM REQUIREMENTS

Although it was not the purpose of this study to define or otherwise develop or evaluate scientific requirements for the multiple satellites, it was necessary to investigate the interaction of scientific constraints with the system to establish the system requirements. Therefore, the scientific objectives and instrument complement of the Multiple Satellite Program, as specified by Ames Research Center, have been interpreted to define the multiple satellite mission and the related mission requirements. Four basically different mission modes or options for accomplishing the mission were defined and evaluated based upon the implication of scientific instrument requirements, cost, weight, and reliability. Based upon this evaluation, one mission option was specified for which the system requirements and operational sequence were established. The sections which follow describe the results of this science/system requirement investigation.

2.1 SCIENTIFIC OBJECTIVES[1]

"In the vicinity of the Earth the solar wind encounters the magnetosphere which presents an obstacle to its passage similar to that of a blunt body in aerodynamic flow. The interruption of flow creates a shock wave, followed by a shocked gas region and boundary layer around the magnetosphere, with the free surface of the magnetosphere itself determining the detailed flow properties near the boundary or magnetopause. Magnetic and plasma measurements in the subsolar region of the magnetosphere heretofore furnished only sequential single-point types of information regarding

(1) the local velocity of the solar wind,

(2) the magnetic field intensity at any point in this region,

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[1]Quoted statements are from the Specification A-11967, Revision A, NASA Ames Research Center, 1 Sept. 1966.

(3) the derived plasma density, and

(4) an approximation of the temperature of the region around the point at which the measurement was made."

"There are indications, however, that these parameters may not be invariant but rather that they undergo extreme fluctuations due to discontinuities or changes occurring in the co-moving frame attached to the plasma. To better understand these phenomena it is necessary to investigate the situation in a frame of reference which allows separation of time-dependent events from the motion associated with the disturbances being convected or propagated within the plasma. Simultaneous measurements of these phenomena will assist in determination of scale-size, shape, propagation characteristics and time-rate-of-change of such disturbances and will provide a more complete picture of the composition and micro-structure of the magnetosphere environment."

"The scientific purpose, therefore, is the detailed examination of the solar wind, the transition region or magnetosheath, and the magnetosphere itself. The separation of wave motions in the collision-free plasma in space from the convection of instabilities and quasi-stationary structure is of primary concern. Most of the areas of interest can be investigated thoroughly in situ using a plasma probe and magnetometer to make measurements at two or more points simultaneously."

These scientific objectives are translated into a mission description and related mission requirements in the following sections.

2.2 MULTIPLE SATELLITE MISSION AND REQUIREMENTS

The multiple satellite mission can best be described by reference to Figure 1. Shown schematically in the figure is the highly elliptical orbit passing through the transition region. The magnetosphere boundary is shown to be generally between 8 and 12 R_e (Earth radii) with the transition region extending to 12 to 16 R_e where the shock front separates the transition region from interplanetary space. The bulk velocity of the plasma in the interplanetary media is 300 to 700 km/sec. Superimposed on the bulk velocity are disturbances with propagation velocities of 100 to 1000 km/sec. These disturbances

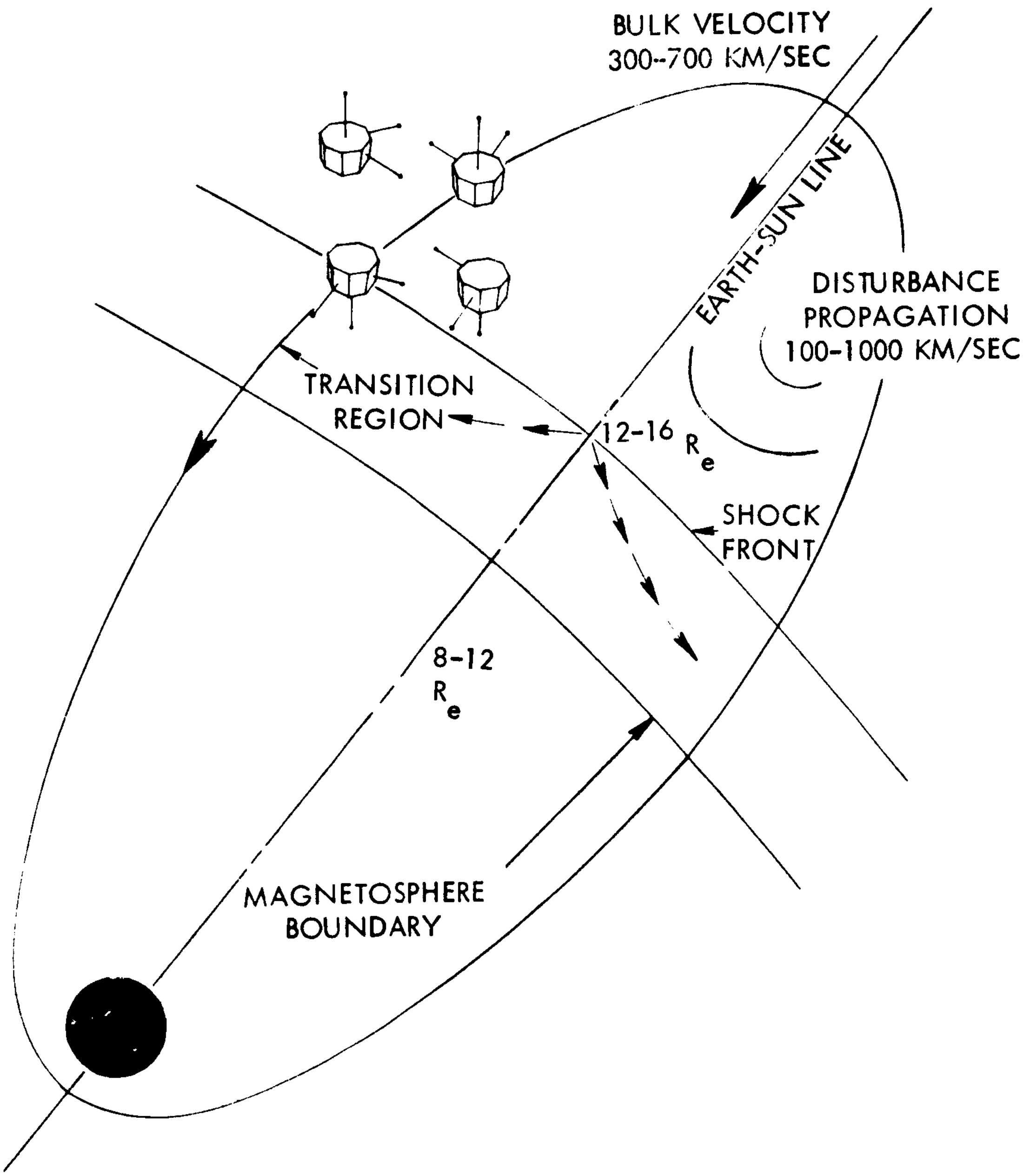


Figure 1. Multiple Satellite Missions

may propagate in any direction. The bulk velocity and disturbance velocities are somewhat lower in the transition region, i.e., the bulk velocities are typically 100 to 200 km/sec in the transition region near the subsolar line.

The multiple satellite mission is comprised of a non-coplanar array of four satellites in a highly elliptical orbit so that data can be acquired in the transition region and near interplanetary space. Figure 1 shows the major axes of the orbit aligned with the Earth-sun line. The satellites should be launched so the apogee and the major axis lead the subsolar region and rotate through it after the satellites have been separated and are operating.

The basic mission requirements are summarized as follows:

a. Experiments - From the scientific goals, which are to understand the structure of the plasma discontinuities and to understand their interaction with the geomagnetic field, the objectives of the experiments are twofold: (1) to measure the propagation direction and speed of solar wind disturbance fronts in the transition region and beyond the shock front, and (2) to derive data permitting temporal and spatial variations in the solar wind and the magnetic field to be separately identified.

b. Data Acquisition and Coverage - Maximum coverage and data acquisition will occur in the transition region and immediately beyond in interplanetary space.

c. Orbit - The multiple satellite orbit must be highly elliptical extending beyond the shock front into near-interplanetary space. A minimum apogee radius of 18 R_e is required with a perigee altitude sufficiently high to assure a one-year life. Launch time and orbit orientation shall be selected to maximize coverage of the subsolar region.

d. Satellite Array - A non-coplanar array of satellites is required to guarantee detection of disturbances propagating in all directions. This requires a minimum of four satellites. The satellites must have a minimum separation of at least 500 to 1000 km in the region of interest in order to detect the disturbances and should not have greater separations than 15,000 km after a period of six months.

e. Satellite Stabilization and Orientation - The satellites will be spin-stabilized at a rate between 50 and 70 rpm. The preferable orientation of the spin axis is perpendicular to the ecliptic plane, to maximize the viewing window coverage of the plasma instruments.

f. Design Requirements - The system design is required (1) to maximize instrument weight, volume and data rate while maintaining system simplicity through the use of off-the-shelf components where possible; (2) to achieve high reliability (0.70 after a period of three months); (3) to evolve low development cost and (4) to achieve a high degree of magnetic cleanliness (0.5 gamma at the magnetometer sensor).

g. Double Satellite Considerations - An alternate mission mode which was briefly considered involves the launching of two of the multiple satellites into a 40 R_e apogee orbit, and then providing appropriate means for separating and orienting them. As previously stated the multiple satellite scientific objectives are best satisfied by a non-coplanar array of four satellites. Therefore, the initial multiple satellite program should consider the four satellite case with later considerations possibly given to the double satellite mission.

The payload capability for placing two satellites in a 40 R_e apogee orbit will be more than adequate, using the Thor series boosters. Based upon the above considerations, no specific design work was completed for the double satellite mission, although nothing in the current design concept would preclude its implementation.

2.3 MAJOR MISSION OPTIONS

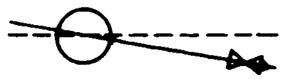
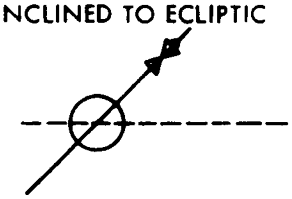
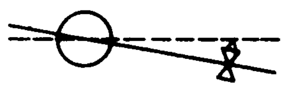
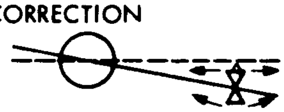
Four major mission options for the Multiple Satellite Program were defined for comparison. Each of the mission options differed in orbit and spin axis orientation, and hence in the design and complexity of the array delivery system and in the satellites themselves. Figure 2 illustrates the differences in the major mission options. The options are:

Option 1 - No spin axis reorientation; orbit plane near the ecliptic.

Option 2 - No spin axis reorientation; orbit plane inclined to the ecliptic.

Option 3 - Spin axis reorientation by pallet attitude control system prior to deployment of the satellites; orbit plane near the ecliptic.

Option 4 - Spin axis reorientation by separate satellite systems after deployment from pallet; satellites provide a post-separation orbit and attitude adjustment capability; orbit plane near the ecliptic.

	PLASMA PROBE VIEW ANGLE	SUBSOLAR COVERAGE	ARRAY CONTROL	REL. AND DEV. COST	WEIGHT
1 ORBIT & SPIN AXIS NEAR ECLIPTIC 	−			+	+
2 ORBIT & SPIN AXIS INCLINED TO ECLIPTIC 		−		+	
3 ORBIT NEAR ECLIPTIC SPIN AXIS 	+	+			
4 ORBIT NEAR ECLIPTIC SPIN AXIS VELOCITY & ATTITUDE CORRECTION 	+	+	+		

NOTE:

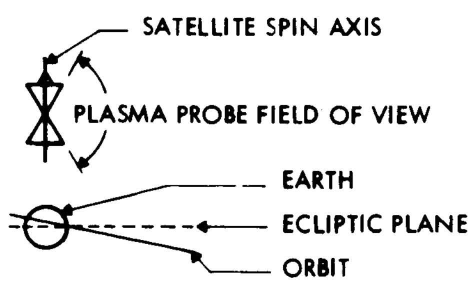


Figure 2. Major Mission Options

In the first and second options, the satellite's spin axis will lie in the orbital plane. For these options the requirement for maximum coverage of the subsolar point then conflicts with the requirement for keeping the sun line within the plasma probe's view field throughout the operational lifetime. Option 1 does not provide the desired plasma probe view field. Option 2 has the orbit plane inclined to the ecliptic resulting in reduced subsolar coverage, but does provide for the possibility of the spin axis being normal to the ecliptic. Both Options 3 and 4 provide the opportunity for maximum coverage of the subsolar region, since the orbit may lie close to the ecliptic. The sun line could be kept in view at all times by orienting the spin axis normal to the ecliptic.

The deployment of the satellites for each of the options differs. The basic differences relate to system complexity and flexibility to control the satellite array. Options 1, 2, and 4 require a very simple pallet having the primary purpose of releasing the satellites at the proper time in the orbit and the spin cycle. Option 3 has a more complicated pallet containing an attitude control system which reorients the pallet/satellite combination normal to the orbit plane before release. In Option 4, the burden of reorientation is placed on the satellites, each of which contains an attitude control system. With minor additions it is possible to provide the satellites of Option 4 with velocity correction capability which allows control of the satellites' array and its growth. With each additional capability provided, however, the system becomes more complex. The functional differences in the systems have been evaluated by comparative block diagrams.[2]

2.4 IMPLICATION OF SCIENCE, RELIABILITY, COST AND WEIGHT ON THE MISSION MODE SELECTION

Mission Option 3, which utilizes a pallet for reorientation and deployment of the satellites, was selected for further analysis and design. The bases for this selection are as follows. Figure 2 indicates positive and negative factors associated with each option in the areas of plasma probe view angle coverage, subsolar coverage, array control, weight, reliability and

[2] See Appendix I

development cost. Both Options 1 and 2 are rejected for scientific reasons; Option 1 provides unacceptable plasma probe coverage and Option 2 provides unacceptable coverage of the subsolar region. These scientific disadvantages outweigh the reliability and cost advantages of these more simpler systems. Option 1 also has a weight advantage over the other systems, but this does not offset the basic requirement that the spin axis be near perpendicular to the ecliptic plane. Option 2 does not have a weight advantage due to the requirement for a polar launch and the subsequent loss in payload weight.[3]

Both Options 3 and 4 can satisfy the scientific requirements.[4] Option 3 has the capability, if careful design is employed, to achieve the required separation without extreme array growth distances.[5] Option 4, however, has the capability of controlling the array growth and, possibly, changing the array by ground command. However, this additional capability results in a more costly, less reliable and heavier system. Therefore, in keeping with the basic goal of the program of providing a simple and reliable system, Option 3 was selected for further study.

2.5 OPERATIONAL SEQUENCE

Table 1 is the operational sequence that is an outgrowth of the orbit and control analyses and systems analysis tasks which are described in the following sections. It is presented here to provide an early understanding of the basic implementation of the multiple satellite mission. The operational sequence has been designed to minimize equipment on the pallet. Figure 3 shows the position in Earth orbit when each event occurs.

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[3]See Appendix I for quantitative comparisons of weight, cost and reliability.

[4]See Appendix II for a discussion of the scientific instrument implications on separation distances, etc.

[5]See Appendix III.

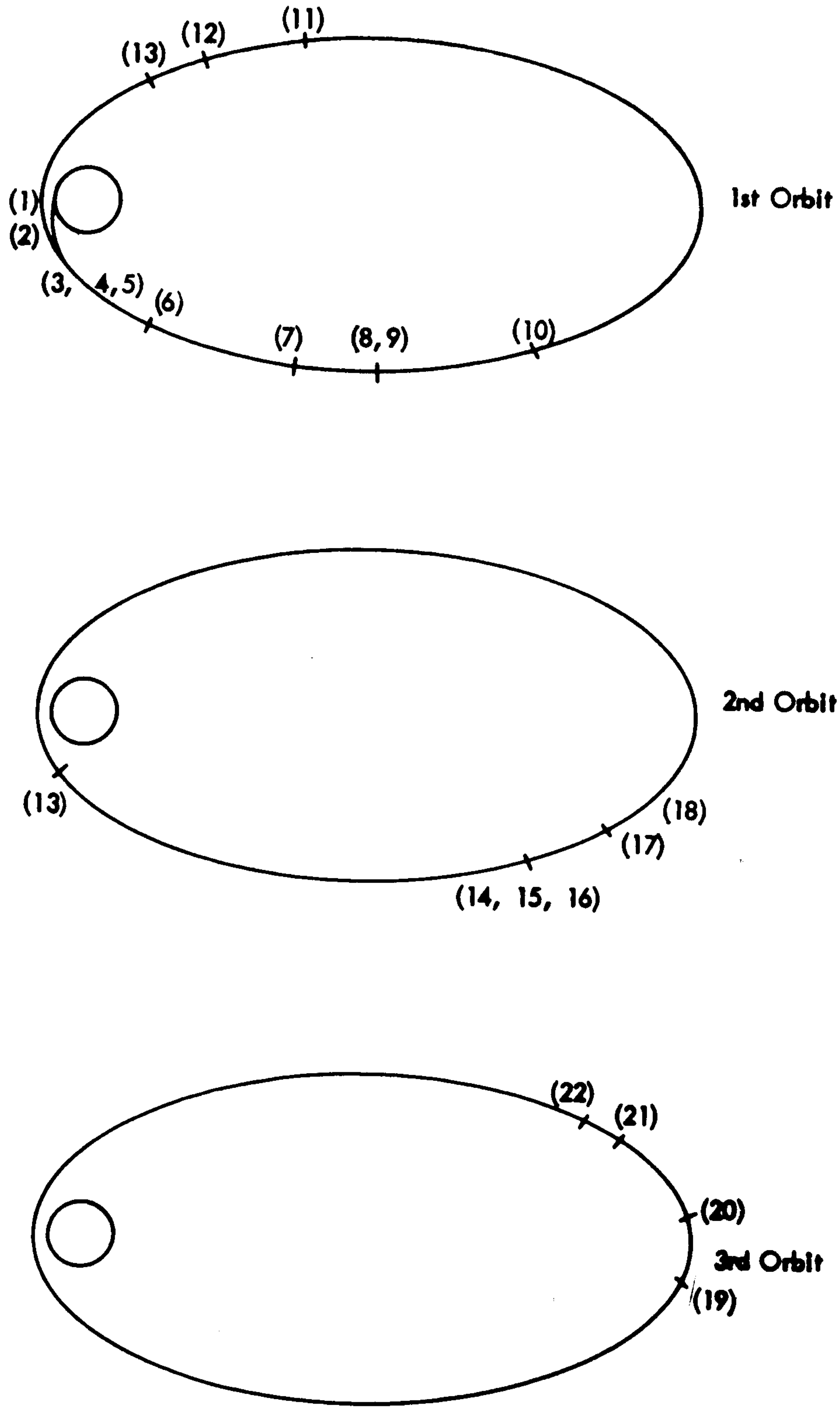

Figure 3. Operational Sequence

Table 1

OPERATIONAL SEQUENCE

1. Satellite and pallet command receivers on.
2. Launch with THORAD into nominal orbit.
3. Pallet antenna deployed using a G-switch release.
4. Pallet beacon on and track spacecraft (G-switch w/command back-up).
5. Separate pallet from spinning third stage.
6. Uncage pallet precession damper (separation switch w/command back-up).
7. Using pallet attitude control system, command ACS mode to orient pallet normal to sun line.
8. Command satellite transmitters on.
9. Using pallet attitude control system, command the appropriate ACS mode to rotate the pallet about the sun line to bring the spin axis toward normal to the orbit plane. The maneuver is completed during the ascending leg of the first orbit so that the fan beam antenna can be oriented for subsequent use in transmission.
10. Command real time data mode, aspect and housekeeping data only. (The instruments are off; transmit blank words.)
11. Command transponder on (transmitter off) intermittently; track and obtain ephemeris data.
12. Using the pallet ACS, trim the pallet attitude.
13. Continue to transmit aspect data during and after attitude maneuvers to verify final orientation.
14. Continue to track intermittently.
15. At 10 hours after perigee passage of the second orbit, command spin-off of satellite pairs to get in-plane normal separation.
16. Command separation of pairs immediately after spin-off. (The pallet function is now complete - beacon off by command w/end-of-life switch override).
17. Uncage satellite precession dampers.
18. Intermittently command all satellite transponders and transmitters on to obtain ephemeris and aspect data.
19. At about 20 hours after perigee passage of the third orbit, command firing of small solid rockets on each satellite to achieve out-of-plane separation. Vary point of pairs separation as a function of ephemeris and aspect data.
20. Deploy magnetometer booms. (The satellites are now deployed and ready for operational use.)

Table 1 (Continued)

OPERATIONAL SEQUENCE

21. Intermittently command transponder and transmitters on to obtain orbit ephemeris and aspect data.
22. Command instruments on.
23. Command continuous real time data mode for about 10 orbits.
24. Command record data mode.
25. Continue to track satellites three times each orbit to up-date ephemeris.

Section 3

ORBIT AND CONTROL CONSIDERATIONS

The launching of the multiple satellite system into a highly elliptical orbit and subsequent deployment of the four satellites into a non-coplanar array is a highly complex maneuver. A number of orbit and control analyses have been performed to verify the technical feasibility of the mission. The best common orbit from which to deploy the satellites has been defined with reference to the scientific objectives. Orbital perturbations have been studied to establish their effect on the common orbit and the adjustments required in the orbit are indicated. The general launch vehicle constraints and launch window freedom for the specified common orbit have been established. The deployment alternatives for the satellites are quite numerous and the details of an optimum deployment have not yet been established. However, the deployment sensitivities as a function of orbital position have been defined and a feasible deployment scheme, which will provide an acceptable non-coplanar array, is presented. Using this deployment scheme the growth of the array over the six-month life has been evaluated by computer simulation.

Control of the pallet and satellites has been analyzed and a system is specified. The pallet attitude reorientation system is described and the expected drift of the satellite's spin axis is computed.

3.1 SELECTION OF COMMON ORBIT PARAMETERS

The multiple satellites will be injected into orbit by a single booster and subsequently separated by small velocity increments to achieve the desired array of intersatellite separation distances. The deployment velocity increments will be quite small in comparison to the orbital velocity and the intersatellite separation distances will be small relative to the dimensions of the orbit. Hence, a logical division can be made between: (a) the "common" orbit, and (b) orbital differences between the satellites resulting from the deployment and giving rise to the separation distances. For the present deployment concept, the pallet receives no velocity increment during the separation and deployment of the satellites. Also, the velocity increments applied to the satellites are differential, i.e., of equal magnitude but of opposite direction for pairs of satellites. Therefore, the common orbit can be taken to be the pallet's orbit. Analyses of the common orbit are summarized below[6].

3.1.1 APOGEE RADIUS AND PERIGEE ALTITUDE

Considerations relating to the initial values of the common orbit parameters can be divided between

a. selection of the orbital dimensions, and

b. selection of the orbit orientation.

The selection of the orbit dimensions involves the selection of apogee radius and perigee attitude.

The main considerations affecting the selection of apogee radius are:

a. Scientific objectives

b. Satellite separation distance array problems

c. Apogee radius dispersion

d. Payload capability

e. Efficient utilization of time in orbit

The scientific objectives impose a lower bound on apogee radius. It is desired that data be obtained at radii far enough beyond the transition region to assure the detection and measurement of approaching disturbances in the solar wind. Along the subsolar line the transition region may extend out as far as 16 R_e. Hence, an apogee radius greater than 16 R_e is called for. (Still larger apogee radii are required to obtain data beyond the transition region when the direction of the line of apsides is at a substantial angle to the sunline.)

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[6] See also Appendix IV.

Consideration of the selected deployment scheme indicates that, at least during the first few months in orbit, the most desirable separation distance array conditions will not be obtained in the vicinity of apogee. Good array non-coplanarity conditions will extend to radii somewhat below apogee. Since it is necessary that these conditions be obtained beyond the transition region, the minimum requirements on the apogee radius must be increased. With an apogee radius of 20 R_e, acceptable array condition will be obtained out to about 18 R_e.

Because the third stage of the Thor-Delta booster is not cut-off, a substantial error in velocity magnitude is incurred at injection. This will be reflected in a 3σ dispersion of apogee radius which is estimated to be close to 2 R_e. Since the consequences of an excessively small apogee radius are more serious than the consequences of a larger nominal apogee radius, it is advisable to bias the booster flight to favor the higher apogee altitude. With a nominal apogee radius of 20 R_e, a lower bound of 19 R_e and an upper bound of 23 R_e should be satisfactory and achievable.

The cost of larger apogee radii in decreased payload capability is minor, about 2 pounds/R_e in the 20 R_e altitude range. More significant, perhaps, is the decreased efficiency of time utilization. If the apogee radius is larger than necessary the orbital period will be larger than necessary and the number of orbits from which data is obtained in a given operational lifetime is reduced. Furthermore, the portion of an orbit period spent near apogee, where the data collected is not of the greatest interest, is increased.

No major variations in communications coverage within the contemplated range of apogee radius is expected.

The main considerations involved in the selection of perigee altitude are:

a. Payload capability

b. Differences in orbital decay due to atmospheric drag

c. Maintenance of adequate perigee altitude while subject to orbital perturbation effects

d. Spin axis drift

The trade-off between perigee altitude at injection into orbit and payload capability for the DSV-3K (Thorad/Improved Delta/FW4) booster vehicle, for orbits in the region of interest, is about .09 pounds per km, i.e., an increase in perigee altitude of 200 km would result in a payload loss of about 18 pounds. It is possible to achieve a high perigee altitude for the pallet more economically by injecting into a low perigee altitude and then increasing perigee altitude by applying a velocity increment to the pallet at apogee. Although it appears that the booster payload capability will be adequate without recourse to this complication, this alternative may demand further consideration. In general, though, to conserve payload weight, it is desirable that the perigee altitude at injection be as low as possible consistent with the mission requirements.

The main restriction on lowering perigee altitude arises from the difference in orbital decay due to atmospheric drag resulting from differences in perigee altitude between satellites. Differences in orbital decay lead to differences in orbital period which, in turn, cause a steady increase in the rate of growth of differences in lead time and separation distances along the orbit. It is desired that the growth of separation distances be restricted to less than 15,000 km in the first six months. Since, adherence to this restriction imposes serious constraints on the accuracy of the satellites' deployment, it is felt that the growth due to differences in orbital decay should be kept small relative to the total allowable growth.

For the current deployment scheme, a difference in perigee altitude of about 240 km will arise initially as a result of the deployment velocity increments. As indicated in Figure 4 with this perigee altitude difference the lowest satellite must have a perigee altitude equal to or above 370 km to restrict the six-month growth of separation distance to about 3750 km.

The only orbital perturbation effect that has a major influence on perigee altitude is that due to solar attraction. This effect causes perigee altitude to oscillate with a 1/2-year period. Depending upon the phase of the oscillation at the outset, the perturbation can cause either an increase or decrease in perigee altitude. However, since the starting phase of the oscillation depends upon the initial direction of the sunline relative to the orbit to maximize coverage of the sub-solar line, the orientation of the orbit will be

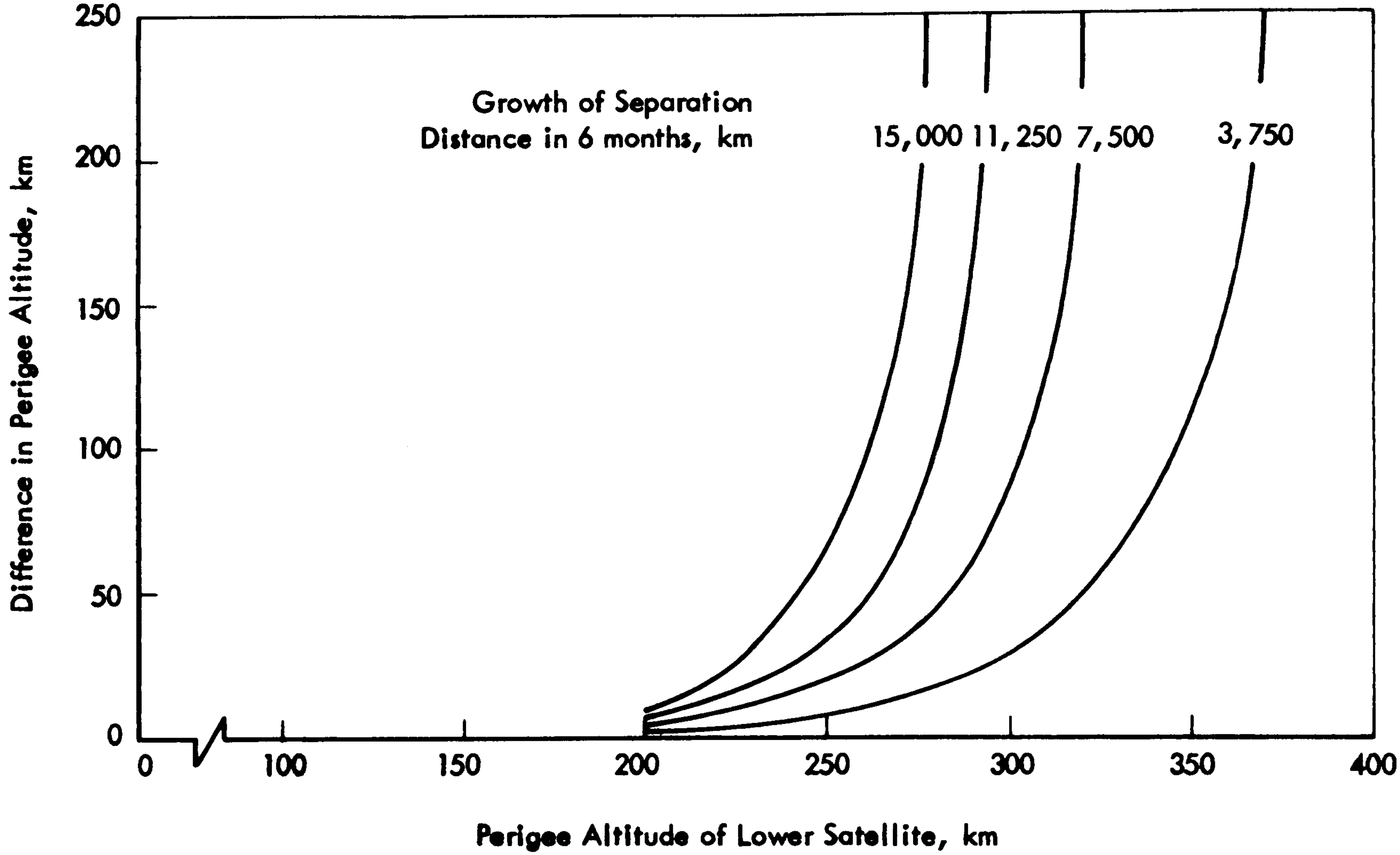

Figure 4. Effect of Perigee Altitude on Growth of Separation Distances

chosen so that an initial angle of from 25 to 45 degrees between the line of apsides and the sunline is obtained. Under these conditions, the phase of the solar perturbation will be adverse. In the worst case, a -90 degree phase-angle will be obtained. This will cause a drop of about 260 km in perigee altitude during the first 45 days in orbit.

Adding 260 km to allow for the solar perturbation and 120 km (half the total difference due to deployment) to the 370 km minimum altitude, an initial perigee altitude of 750 km for the pallet is required. Expressed sequentially for an initial perigee altitude of 750 km for the pallet immediately after deployment, the satellites' perigee altitudes will range from 630 to 870 km; after 45 days, due mainly to the solar perturbation, the satellites' perigee altitude range will drop to values from 370 to 610 km.

3.1.2 ORIENTATION

The most important considerations regarding the orientation of the orbit relate to coverage of the sub-solar line[7]. This coverage depends upon the orientation of the orbit relative to the ecliptic plane and the sunline. As illustrated in Figure 5, the parameters involved are: (a) i_e, the inclination of the orbit plane relative to the ecliptic plane; (b) θ_c, the true anomaly of the line common to the two planes, and (c) s_a, the angle between the sunline and the line of apsides. The former two parameters are subject to relatively small changes due to orbital perturbation effects; the latter changes mainly as a result of the one-degree/day rotation of the sunline.

Since the conditions of most scientific interest are obtained when apogee is in the general direction of the sunline, it is important that the initial value of s_a be such as to provide a suitable lead of the time when the sunline crosses apogee. A nominal initial angle of 35 degree provides a lead time of about 35 days. During the first 70 days in orbit, when the probability of successful collection of data is greatest, the apogee radius would be within 35 degrees of the sunline. To allow for a launch window of reasonable width, a lead time tolerance of $\pm$ 10 days, i.e., a tolerance of $\pm$10 degrees in s_a, has been assumed.

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[7]See Appendix V for detailed analysis.

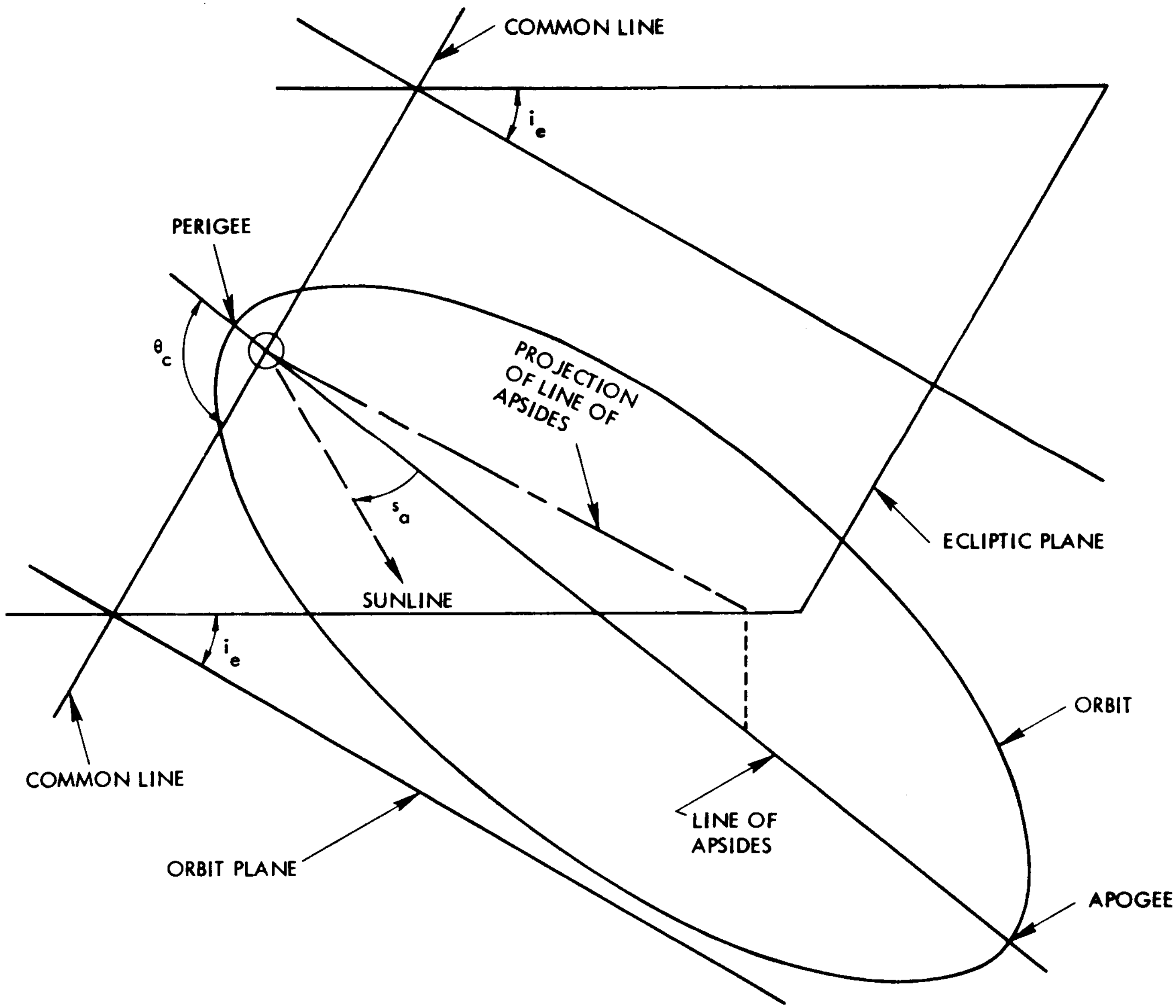


Figure 5. Relationship Between the Ecliptic and Orbital Planes

For scientific purposes, it is desired that the satellites be separated as little as possible from the ecliptic plane, particularly in the section of the orbit from 8 to 16 R_e. The attainment of this objective depends upon the values of i_e and θ_c. If a zero value of i_e could be obtained, the orbit would lie entirely in the ecliptic plane. However, since a dogleg in the boost trajectory is to be avoided, the minimum inclination (relative to the ecliptic that can be obtained with a launch from ETR) is slightly greater than 5 degrees. With i_e not equal to 0, the distribution of distances from the ecliptic depends upon the value of θ_c. The two points that lie on the common line, i.e., the points having a true anomaly equal to θ_c and to θ_c + 180 degrees, will lie in the ecliptic plane. For other points, the distance from the ecliptic plane is proportional to the distance from the common line. Figure 6 shows the distribution of distance from the ecliptic plane for various values of θ_c.

With the present deployment scheme, a better array of separation distances will be obtained on the descending leg of the orbit than on the ascending leg. Hence, the value of θ_c should be chosen insofar as possible to favor the descending leg of the orbit. For this purpose a value of θ_c near 30 degrees would be most desirable. Because of launch window constraints, it will probably be necessary to accept a fairly wide range of values. However, it should be possible to keep θ_c in the range of 0 to 60 degrees.

3.1.3 NOMINAL ORBIT SUMMARY

The nominal orbit selected is shown in Figure 7. It has the following characteristics:

<u>Dimensions</u>

Apogee radius - 20 Earth radii +3, -1
Perigee altitude - 750 km, initial

<u>Period</u>

48 hours, nominally

<u>Orientation</u>

Inclination - 5 degrees, + 1
True anomaly of common line: $0 < \theta_c < 60$ deg.

<u>Lead Time</u>

25 to 45 days

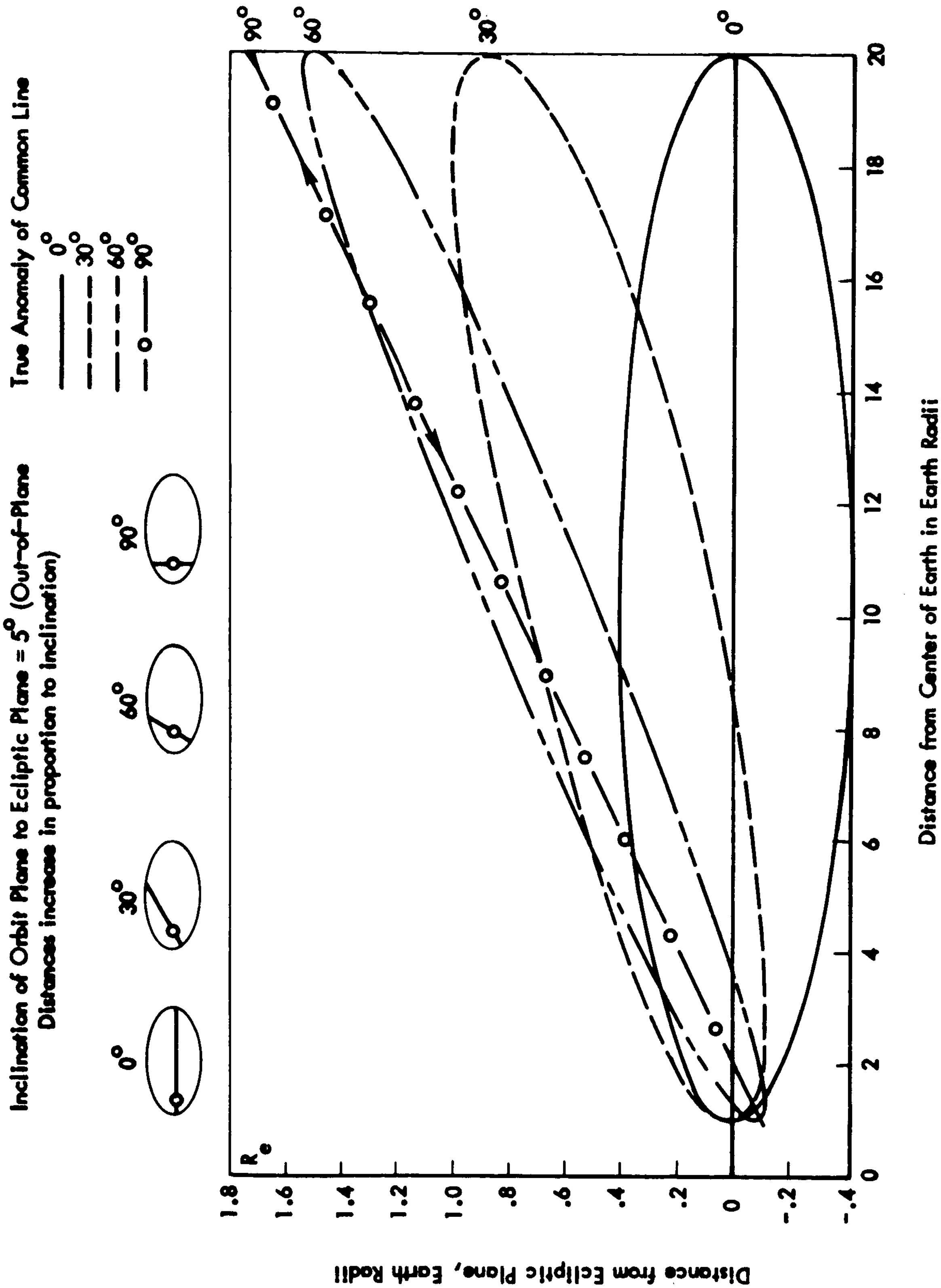
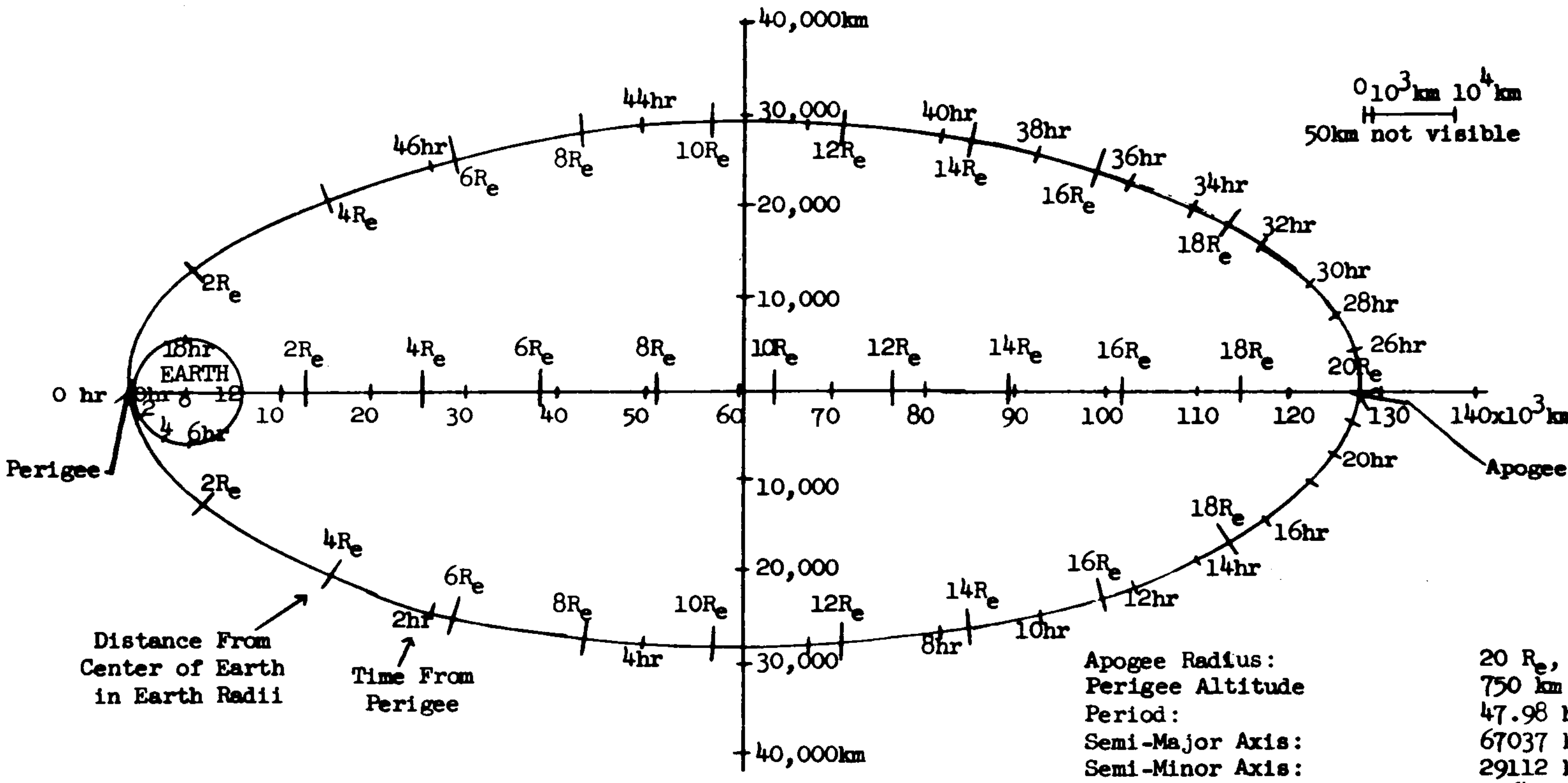


Figure 6. Effect of True Anomaly of Common Line on Out-of-Plane Distances



Apogee Radius:	20 R_e, 127,424 km
Perigee Altitude	750 km
Period:	47.98 hrs
Semi-Major Axis:	67037 km
Semi-Minor Axis:	29112 km
Semi-Latus Rectum:	12642 km
Velocity at Perigee:	10,673 meters/sec
Velocity at Semi-Minor Axis:	2472.5 meters/sec
Velocity at Apogee:	557.1 meters/sec

Figure 7. Multiple Satellite Reference Orbit

3.2 ORBITAL PERTURBATION EFFECTS

The main orbit perturbation effects are those due to:

a. Atmospheric drag

b. Solar and lunar attractions

c. Earth oblateness

The primary effect of the atmospheric drag is to reduce orbital period. However, the greatest significance of this effect relates to the differences between satellites rather than to absolute effect on the common orbit. With a common orbit perigee altitude of 750 km the effect on the pallet's orbital period is very slight; in the order of .001 percent in six months, and the effect on the orbit dimensions is negligible. Neither does atmospheric drag significantly effect the orientation of the orbit. The angle between the satellite's spin axis and the orbital velocity vector will remain close to 90 degrees and the geometry of the satellite is such that no lift forces of any consequence are anticipated.

Solar and lunar attractions will produce cyclical effects on orbital velocity, causing some speed-up and slow-down over and above the Keplerian variation of orbital velocity as the orbit is traversed. However, to first order the net effect of these perturbations on orbital period vanishes. The paramount effect is on the perigee altitude and, since the net change in period is slight, this is accompanied by a nearly opposite change in apogee attitude. The effects can be described in terms of: (a) an oscillatory component, which for the solar perturbation has a period of one-half-year and for the lunar perturbation has a period of one-half-month; and (b) a secular component. The amplitude of the oscillatory component of solar perturbation effect on perigee altitude is about 260 km while the amplitude of the oscillatory lunar component is about 50 km. The maximum values of the secular contributions after six months in orbit is about 6 km for the solar perturbation and 50 km for the lunar perturbation. Thus, the oscillatory solar contribution is the dominant factor. The phase of this contribution, which determines whether perigee altitude will initially rise or fall, expressed in degrees, is approximately equal to minus twice the lead time of passage of the sunline over apogee in days. Thus the case of the nominal lead time of 35 days corresponds to a phase angle of -70 degrees. This results in a drop in perigee altitude of about 175 km as shown in Figure 8. With a 45-day lead time, the drop in perigee altitude would be equal to a full amplitude of the perigee oscillation caused by the solar perturbation, i.e., about 260 km.

The effects of solar and lunar perturbations on orbital orientation are small in comparison to the effects of earth oblateness. In a six-month period, earth oblateness effects will produce a nodal regression of about 12.5 degrees and an apsidal precession of about 22 degrees. For nominal launch conditions this would result in an increase in inclination of about 1.5 degrees and a decrease in θ_c of about 35 degrees in the six-month period. The earth oblateness has negligible effect on the orbital period or dimensions.

3.3 LAUNCH VEHICLE AND LAUNCH WINDOW CONSIDERATIONS

Table 2 lists the payload capabilities[8] and estimated launch costs for the versions of the Thor-Delta series of booster vehicles applicable to the Multiple Satellite Program. The DSV-3E is the best choice among the existing versions. The DSV-3K and DSV-3L are future versions which are very likely to be current by the time of the multiple satellite launch. Since the payload capability attained with the DSV-3K version appears to be adequate, the availability of this version is most assured and the application of this version to the multiple satellite mission is most direct, it has been chosen as the primary booster vehicle candidate for the present study.

The minimum inclination to the ecliptic plane of about 5 degrees is obtained by a due-east launch from ETR at the time of day when the launch site is closest to the ecliptic plane. The angle between the launch site and perigee, φ_p, and the day of the year, fixes the orientation of the orbit relative to the sunline and thus the lead time before the sunline passes over the apogee radius. If the booster flies a direct ascent trajectory designed for maximum payload capability the value of φ_p will be fixed. Assuming a value of φ_p of 30 degrees, the value of θ_c would be about 60 degrees and the desired nominal lead time of 35 days would be obtained with a launch day of about 15 December[9].

[8] See Appendix VI for detailed launch vehicle analysis.

[9] The launch window is discussed in detail in Appendix VII.

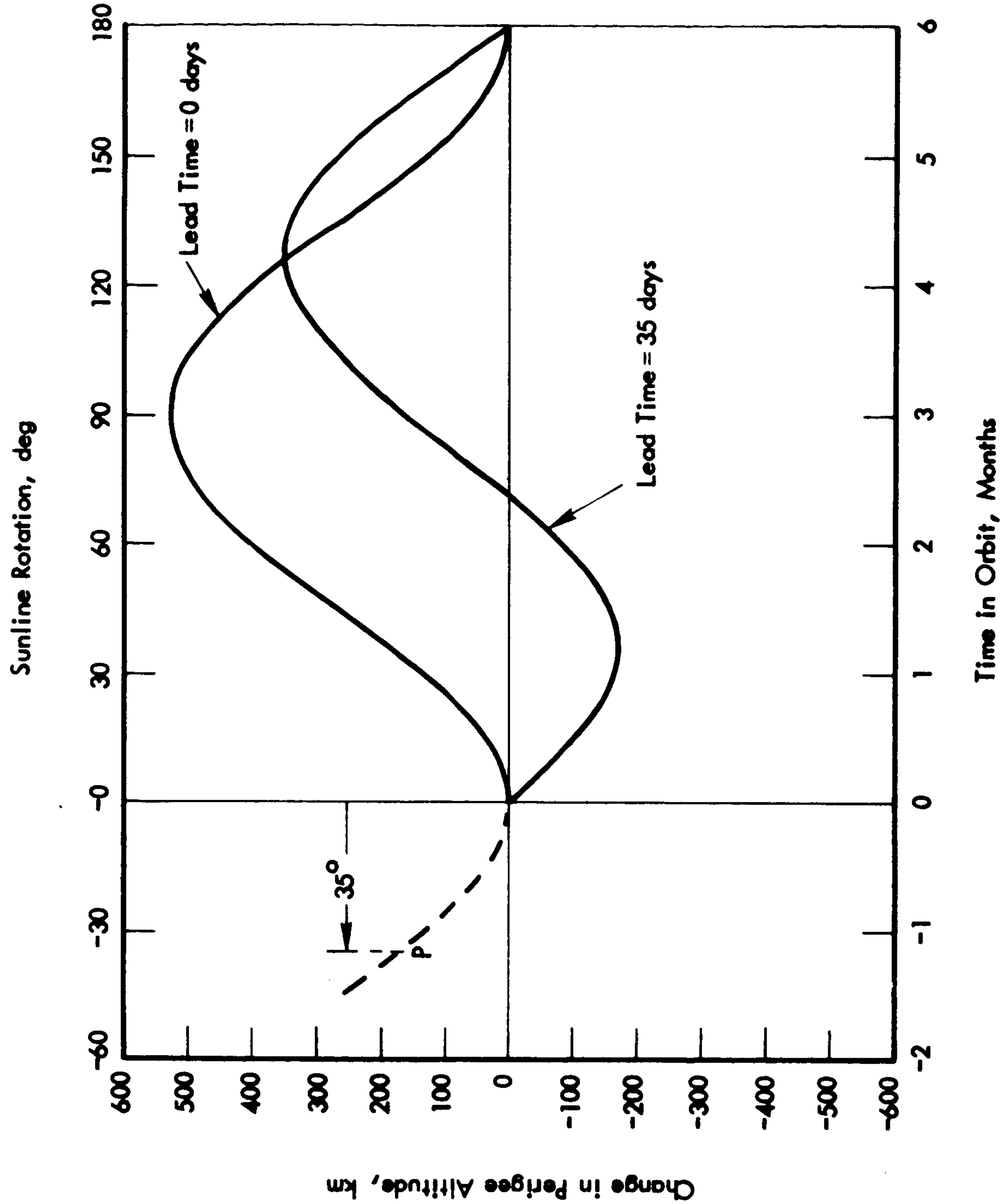


Figure 8. Effect of Solar Perturbation on Perigee Altitude

Table 2

DATA ON BOOSTER VEHICLE PAYLOAD CAPABILITY AND COSTS

Vehicle Designation[1]	DSV-3E	DSV-3K	DSV-3L
1st Stage	TAT	Thorad	Thorad
2nd Stage	Improved Delta	Improved Delta	Improved Delta
3rd Stage	FW-4	FW-4	TE-364
Payload Capability			
For East Launch from ETR			
Into 20 R_e x 280 km orbit	318	444	574 lbs
Into 20 R_e x 280 km orbit - less weight required for ΔV to raise pallet's h_p to 750 km.	314[2]	438[2]	566[2] lbs
Into 20 R_e x 750 km orbit	275	405	480 lbs
Cost Per Launch	\$3.66 million	\$4.06[3] million	\$4.15[3] million

(1) In accordance with designations used in NASA "Certified Launch Vehicle Data"

(2) Based upon use of a solid rocket with I_{sp} = 250 sec and mass fraction .5

(3) Includes amortization costs at current rates

With a fixed value of φ_p, a launch window in the order of $\pm$ 20 days, i.e., a launch window from about 25 November to 5 January, can be obtained by: (a) accepting a range of values of lead time of $\pm$ 10 days, and (b) departing from the nominal launch azimuth and time of day at the expense of some increase in the inclination of the orbit to the ecliptic.

In addition to reducing the deviation from the nominal lead time, departures from nominal azimuth and launch-time-of-day can be used to shift θ_c to more desirable values, i.e., lower values. Since the nominal launch azimuth and launch-time-of-day minimize inclination, departure from nominal launch condition produces second order effects on inclination. Thus, small departures from nominal conditions produce very small increases in inclination, but these increases grow rapidly as the departures from the nominal conditions increase. Since the purpose of shifting the value of θ_c is to get the orbit sections with the best array closer to the ecliptic plane, and increases in inclination tend to defeat this purpose, the degree of control of θ_c that can be gained in this way is limited. Also limiting control of θ_c is the fact that, depending upon the launch day, the requirements on the shifts in launch time and azimuth with respect to maintaining the lead time within allowable limits can conflict with the requirements for the desired shift in θ_c. Detail consideration of these launch window effects indicates that the $\pm$ 20-day window can be obtained, while keeping lead time within 25 to 45 days, the inclination to the ecliptic below six degrees, and obtaining a favorable shift in θ_c in the order of 20 degrees.

Further widening of the launch window requires variation of the value of φ_p. This can be obtained: (a) without restart of the second stage through extension of the coast period between second-stage cut-off and third-stage ignition, and (b) with restart of the second stage by injection into a low-altitude parking orbit at first cut-off of the second stage. The widening of the launch window obtainable with the first approach is limited by the decrease in payload capability that is entailed. It is felt that an increase in φ_p to about 60 degrees could probably be obtained with a modest cost in payload capability. This would double the launch window which would then extend from about 25 November to about 15 February.

At present the three-stage Thor-Delta vehicles are not provided with restart capability. Second-stage restart has been implemented and flown by two-stage versions of the vehicle. This implementation presumably could be adapted to the three-stage versions. In this case, a value of φ_p of about 200 degrees could be obtained; this limit is imposed by the cold-gas supply presently available for the second-stage ACS. The launch window would then be opened to about one-half year, i.e., from 25 November to 15 July. However, in addition to the uncertainty of its availability, use of the restart capability would entail a reduction in reliability and a degradation of guidance accuracy.

The foregoing launch window estimates are predicated upon the ability of the booster vehicle to lift off within a few minutes of the assigned launch time and to accept small changes in launch azimuth from day-to-day while maintaining launch readiness. It is assumed that step changes in φ_p, which entail autopilot reprogramming, will require some downtime and would be made at infrequent intervals, e.g., after 30 days on stand.

Normally, i.e., without restart, the only guidance error of major significance is expected to be the error in velocity magnitude at injection. As previously mentioned, this error is expected to produce a 3σ dispersion of apogee altitude in the order of 2 R_e. Errors in orbit orientation are expected to be in the order of 1 degree, 3σ. The direction of the spin axis at injection will nominally be close to the direction of the velocity vector at perigee. The angle between the velocity vector at perigee and the sunline will range from 45 to 65 degrees depending on lead time. Due to guidance errors, a maximum uncertainty in the initial direction of the spin axis in the order of 5 to 10 degrees is anticipated. Thus, the initial angle between the spin axis and the sunline may range from 35 to 75 degrees.

3.4 DEPLOYMENT SENSITIVITIES

In order to fully achieve the scientific objectives, capability is required for (a) measuring the vector direction and speed of propagation of disturbance fronts, and (b) measuring the gradient (spatial derivatives) of physical variables in three dimensions, a non-coplanar array of four satellites is necessary. The desired array is to be obtained by applying small velocity increments during or subsequent to the separation of the satellites from the pallet[10]. In addition to these deployment velocity increments, differences in the satellites' orbits may be caused by differences in the forces acting on the satellites while in orbit, i.e., differences in orbital perturbation effects. However, for the order of magnitude of the intersatellite separation distances contemplated, the differential orbital perturbations are small and require considerable time before they produce significant changes in the separation distances. Hence, the initial array of separation distances can be attributed almost entirely to the deployment velocity increments; and the history of the separation distances, at least for the most important early portion of the operational lifetime, can be expected to be nearly that which would be obtained for purely Keplerian orbits. Hence, Keplerian orbit calculations have been used to evaluate the sensitivities of the separation distance to deployment velocity increments. These sensitivities serve to guide the selection of the deployment scheme, which is subsequently verified by computer simulation with perturbations[11].

It is convenient to resolve the deployment velocity increments into components: ΔV_T, in the direction of the tangent to the orbit; ΔV_{IN}, in the direction in the orbital plane, normal to the tangent to the orbit; and ΔV_O, in the direction of the normal to the orbit. Similarly, the separation distance can be resolved into components: D_T, D_{IN}, D_O. As illustrated in Figure 9, this coordinate system is localized and rotates when the orbital reference point is moved. The velocity components are referenced to the point of application of the velocity increment. The separation distance components are referenced to the coordinate frame at the point where the separation distances are measured.

The effects of deployment velocity increments on the differences in satellite orbits can be divided into the following three categories:

a. Lead time effects

b. Geometric in-plane effects

c. Geometric out-of-plane effects

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[10] The reference orbit used in the deployment analysis is described in Appendix VIII.

[11] The deployment sensitivities are described in more detail in Appendix IX.

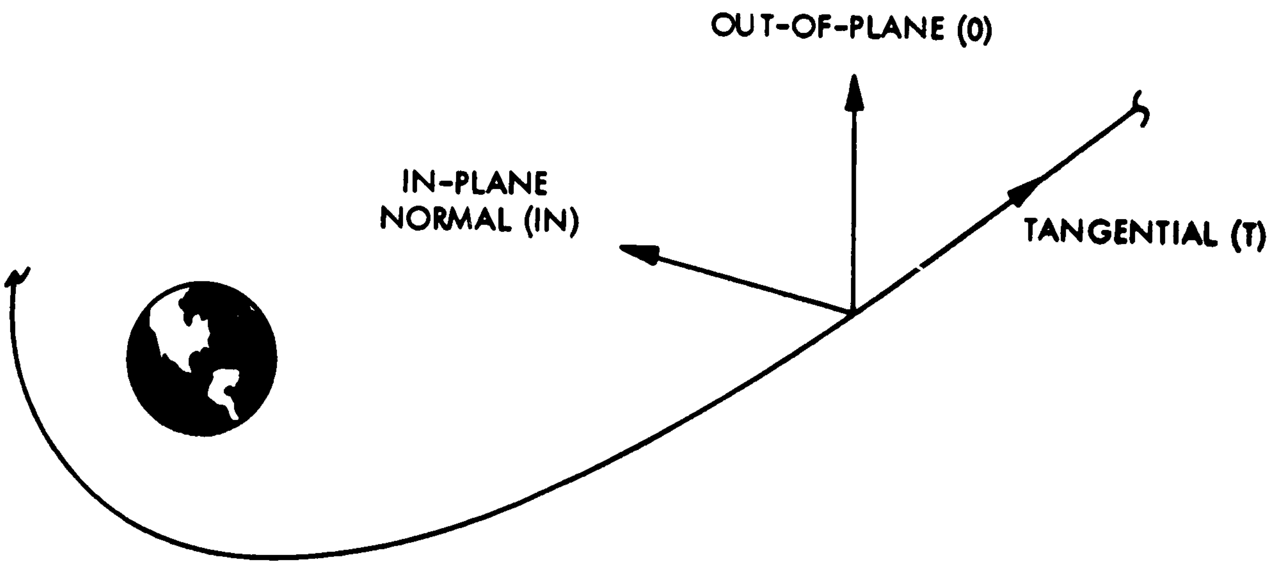


Figure 9. Deployment Coordinate System

Lead time effects involve no differences in the geometry of the orbit. The deployed satellite is displaced relative to the pallet orbit only because it passes a given point at a different time. As illustrated in Figure 10, the separation distance is in the direction of the orbital velocity vector, i.e., in the tangential direction, and rotates as the satellites traverse the orbit. Also, for a fixed lead time, the magnitude of the separation distance varies in proportion to the magnitude of the velocity vector. Thus, the separation distance at perigee is about 20 times greater than at apogee; at the semi-minor axis, i.e., about 10 R_e, the separation distance is about four times larger than at apogee.

In the presence of a difference in orbital periods, the lead time will increase by the difference in period during each orbit, and the separation distance will grow steadily as illustrated in Figure 11. A first order effect on orbital period is obtained only from the tangential component of the deployment velocity increment, ΔV_T. As indicated in Table 3, the orbital period is quite sensitive to this component. Since nominal orbital period is about 2 days, in six months approximately 90 orbits will be traversed. In this time an initial difference in orbital period of .04 percent will produce a lead time difference of 3.6 percent of an orbital period; at 10 R_e, the separation distance, D_T, along the orbit will then be about 15,000 km. Hence, a .04 percent difference in orbital period between satellites is about the maximum that can be tolerated. From Table 3 it is seen that to meet this requirement, the tangential component of the velocity increment must be kept very small - generally below one meter/sec.

As illustrated in Figure 12, geometric in-plane effects include:

a. Change in apogee and perigee altitudes

b. Rotation of the line of apsides

c. Changes in semi-minor axis

These effects are produced only by the in-plane components of the velocity increment, ΔV_T and ΔV_{IN}. Because of the restriction on the magnitude of ΔV_T that is imposed to avoid an excessive growth of tangential separation distance, the geometric effects due to ΔV_T will be comparatively slight. Since the ΔV_{IN} component has very little effect on orbital period, the change in the major axis of the orbit which is directly related to orbital period, will be very small. Hence, the change in perigee altitude will be very nearly equal and opposite to the change in apogee radius.

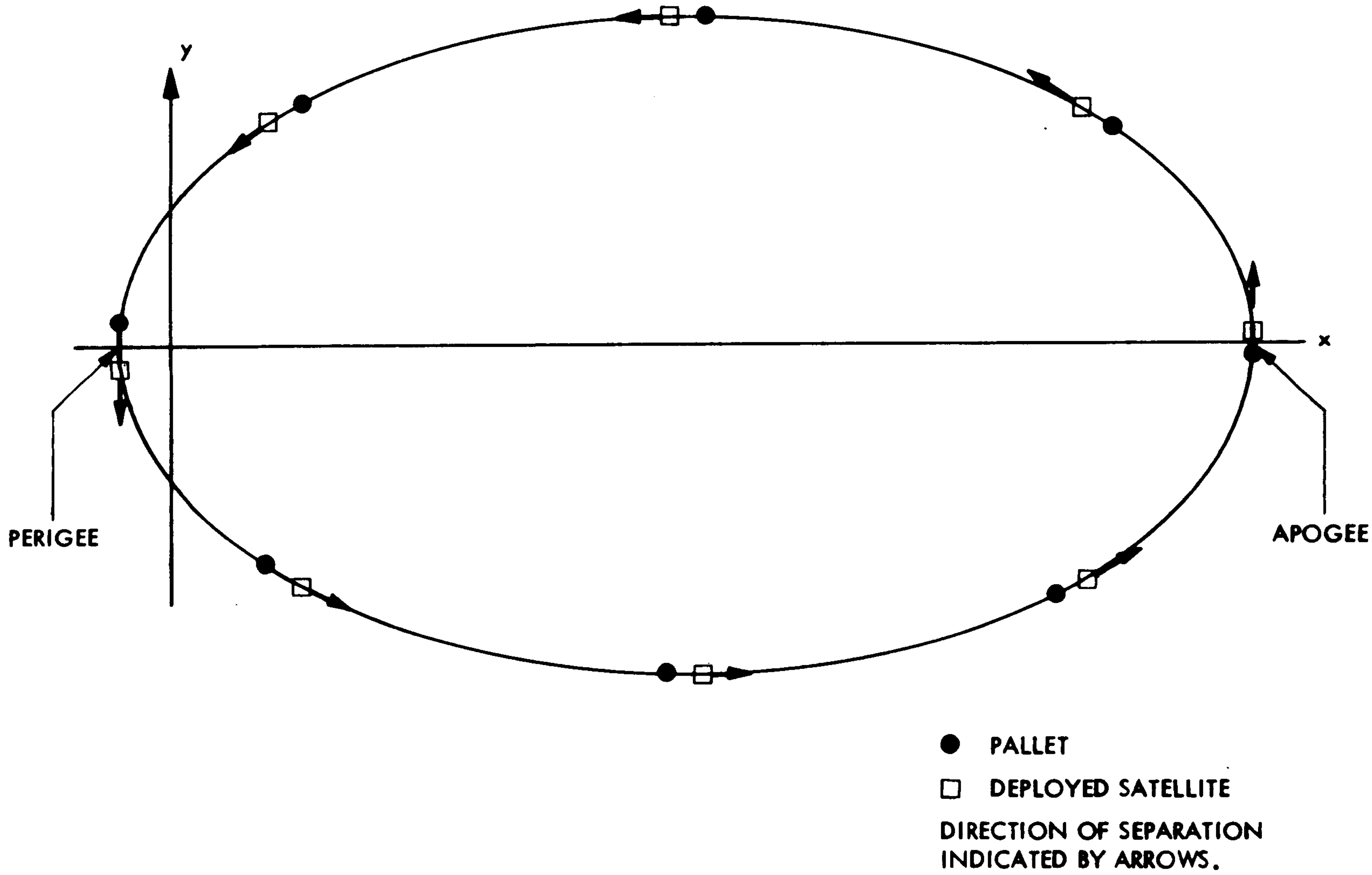


Figure 10. Effect of Lead Time on Satellite Separation Distance

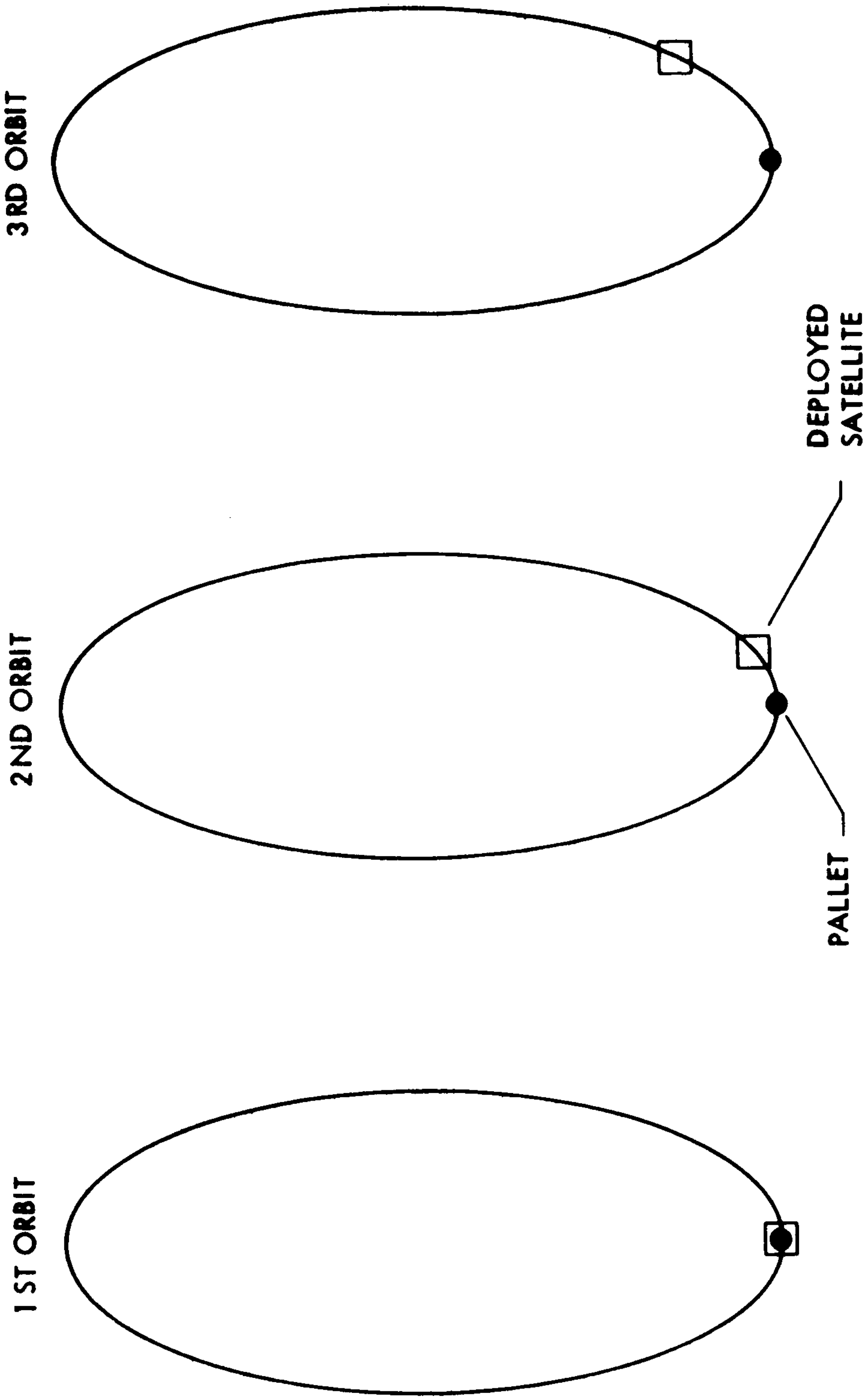


Figure 11. Effect of Difference in Period on Lead Time

Table 3

IN-PLANE LEAD TIME SENSITIVITIES

Position in Reference Orbit		Effect of .1% of Orbit Period Lead Time on Separation Distance, D_T (km)	% Change in Orbital Period due to ΔV_T = 1 meter/sec
Time From Perigee, hrs	Distance From Earth, R_e		
0	1.044	1843.6	.5447
.5	2.147	1249.8	
1.0	3.492	944.5	.2785
1.5	4.666	789.3	
2.0	5.707	690.7	.2036
2.5	6.646	620.1	
3.0	7.506	565.8	.1668
3.5	8.300	522.0	
4.0	9.039	485.5	.1432
4.5	9.730	454.3	
5.0	10.379	427.1	.1260
6.0	11.568	381.4	.1125
7.0	12.632	343.9	.1015
8.0	13.590	312.1	.0922
9.0	14.455	284.5	.0841
10.0	15.238	260.1	.0769
11.0	15.946	237.7	.0751
12.0	16.584	218.6	.0647
13.0	17.159	200.6	.0594
14.0	17.673	184.1	.0546
15.0	18.131	169.0	.0501
16.0	18.535	155.2	.0461
17.0	18.886	142.6	.0423
18.0	19.187	131.2	.0390
19.0	19.440	121.2	.0360
20.0	19.645	112.6	.0334
21.0	19.804	105.7	.0313
22.0	19.916	100.5	.0297
23.0	19.983	97.3	.0286
24.0	20.000	96.2	.0282

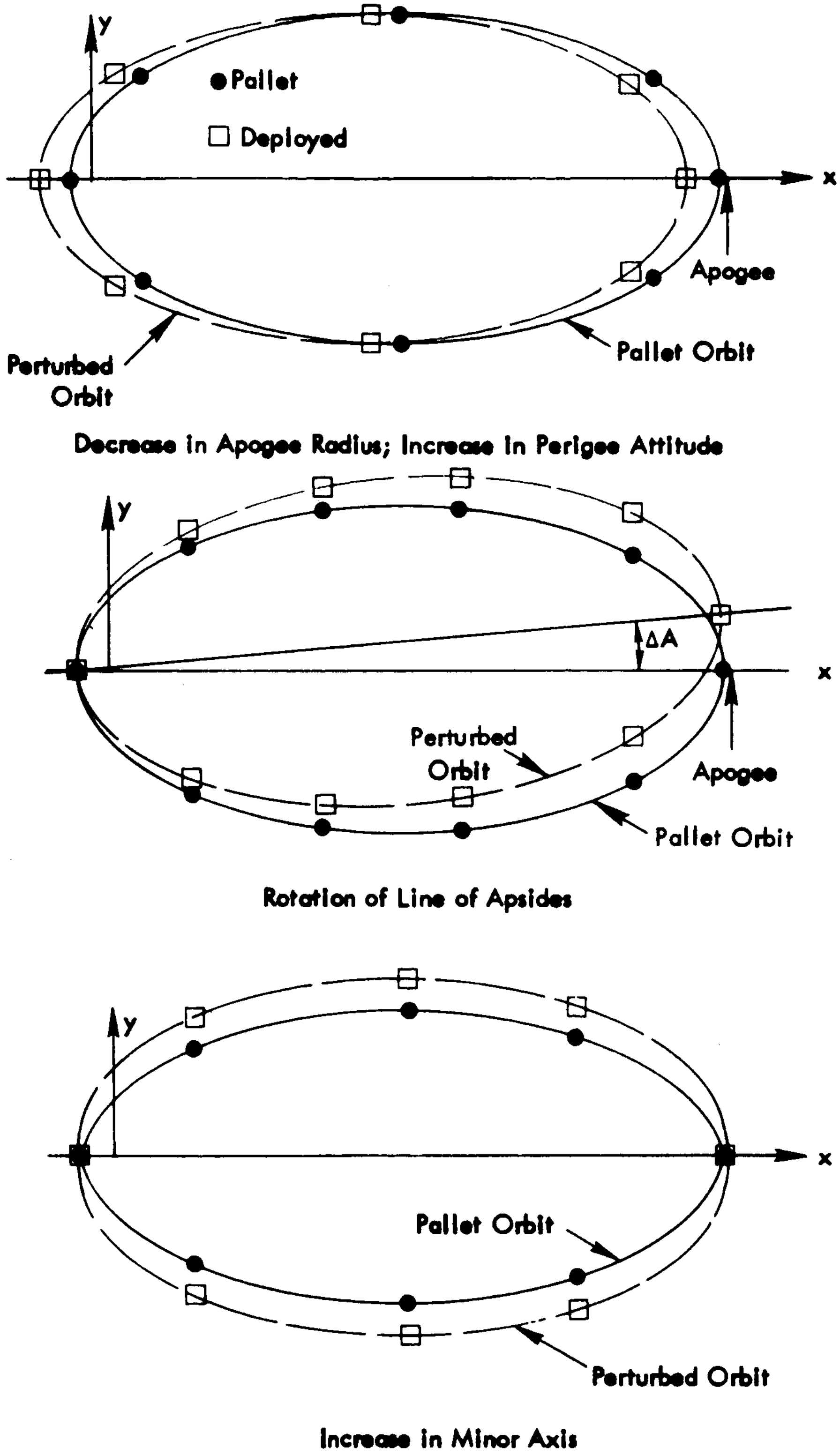


Figure 12. Geometric In-Plane Orbital Effects

In the region of most interest, i.e., 8 to 16 R_e, changes in apogee and perigee altitudes tend to produce mainly tangential separation distances. Minor axis changes and rotation of the line of apsides tend to produce mainly in-plane normal separation distances; generally, the larger contribution is made by the changes in minor axis. Sensitivities of the geometric in-plane effects to the velocity increments are given in Table 4.

It should be noted that the foregoing effects always combine such that the point at which the velocity increment is applied remains on the new orbit. This is illustrated in Figure 13, which shows a typical distribution of the in-plane normal separation distances. It is seen that the largest separation distances are obtained on the leg of the orbit opposite to that on which the velocity increment is applied.

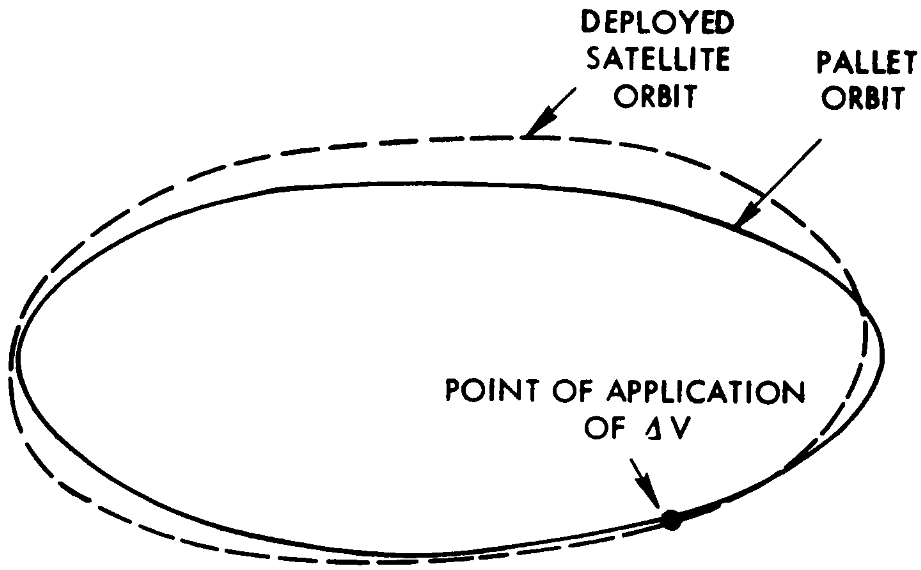
Separation distances in the direction of the normal to the orbit, i.e., out-of-plane separation distances, D_O, are obtained as a result (and only as a result) of the out-of-plane component of the deployment velocity increment, ΔV_O. This component of the velocity increment produces a rotation of the orbital plane about the radius, i.e., the position vector, at the time of application of the velocity increment. That is to say, the true anomaly at the time of deployment is the true anomaly of the line common to the deployed and the pallet orbital planes. To a first order, ΔV_O produces no other changes in the orbit either in geometry or lead time. The relationship between the deployed and the pallet orbits is precisely analogous to the relationship of the pallet orbit to the ecliptic plane; the out-of-plane separation distances depend upon the inclination between the planes and the true anomaly of the common line in the same way as distance from the ecliptic depends upon the corresponding parameters.

As indicated in Table 5, the sensitivity of the inclination of the deployed orbit relative to the pallet orbit increases as the point of application approaches apogee. The sensitivity of out-of-plane separation distance depends upon the point at which the distance is measured. Figure 14 shows a typical distribution of out-of-plane separation distances.

Table 4

GEOMETRIC IN-PLANE SENSITIVITIES

Point of Application of ΔV in Reference Orbit		Tangential Velocity Increment Δv_T = 1 meter/sec				In-Plane Normal Velocity Increment ΔV_{IN} = 1 meter/sec		
Time From Perigee (Hours)	Distance From the Center of the Earth (R_e)	Change in Apogee Radius (km)	Change in Perigee Altitude (km)	Change in Semi-Minor Axis (km)	Rotation of the Line of Apsides (10^{-3} deg)	Change in Perigee Altitude* (km)	Change in Semi-Minor Axis (km)	Rotation of the Line of Apsides (10^{-3} deg)
0	1.044	489.67	0.00	55.9	-0.01	0.02	0.0	11.33
2	5.709	180.14	3.02	27.2	21.89	6.29	14.5	22.47
4	9.046	124.18	4.65	24.4	22.60	10.24	23.6	23.96
6	11.581	95.18	6.09	24.2	22.63	13.12	30.2	22.92
8	13.610	75.42	7.52	25.1	22.41	15.28	35.2	20.52
10	15.267	60.12	9.12	26.8	22.02	16.82	38.7	17.29
12	16.624	47.36	10.87	29.2	21.33	17.38	40.0	13.51
14	17.725	36.17	12.94	32.4	20.34	18.10	41.7	9.18
16	18.599	26.07	15.38	36.6	18.80	17.57	40.5	4.47
18	19.267	16.86	18.22	41.8	16.47	15.94	36.7	-0.53
20	19.741	8.79	21.28	47.6	12.87	12.75	29.4	-5.45
22	20.030	2.70	24.00	52.8	7.54	7.62	17.6	-9.42
24	20.140	0.03	25.30	55.4	0.81	0.86	2.0	-11.28
26	20.070	1.72	24.47	53.8	-6.14	-6.12	-14.1	-10.11
28	19.824	7.15	21.98	48.9	-11.85	-11.70	-26.9	-6.50
30	19.394	14.87	18.92	43.1	-15.80	-15.33	-35.3	-1.75
32	18.772	23.85	16.01	37.8	-18.43	-17.29	-39.8	3.28
34	17.947	33.74	13.47	33.3	-20.13	-18.04	-41.5	8.12
36	16.901	44.64	11.32	29.9	-21.22	-17.91	-41.2	12.55
38	15.608	56.97	9.48	27.3	-21.92	-17.10	-39.4	16.50
40	14.025	71.56	7.87	25.4	-22.38	-15.69	-36.1	19.88
42	12.091	90.02	6.40	24.3	-22.61	-13.67	-31.5	22.47
44	9.691	116.12	4.98	24.2	-22.67	-10.97	-25.3	23.96
46	6.586	162.17	3.43	26.1	-22.25	-7.31	-16.8	23.25
48	2.046	339.59	0.10	39.0	-17.14	-1.73	-4.0	15.41



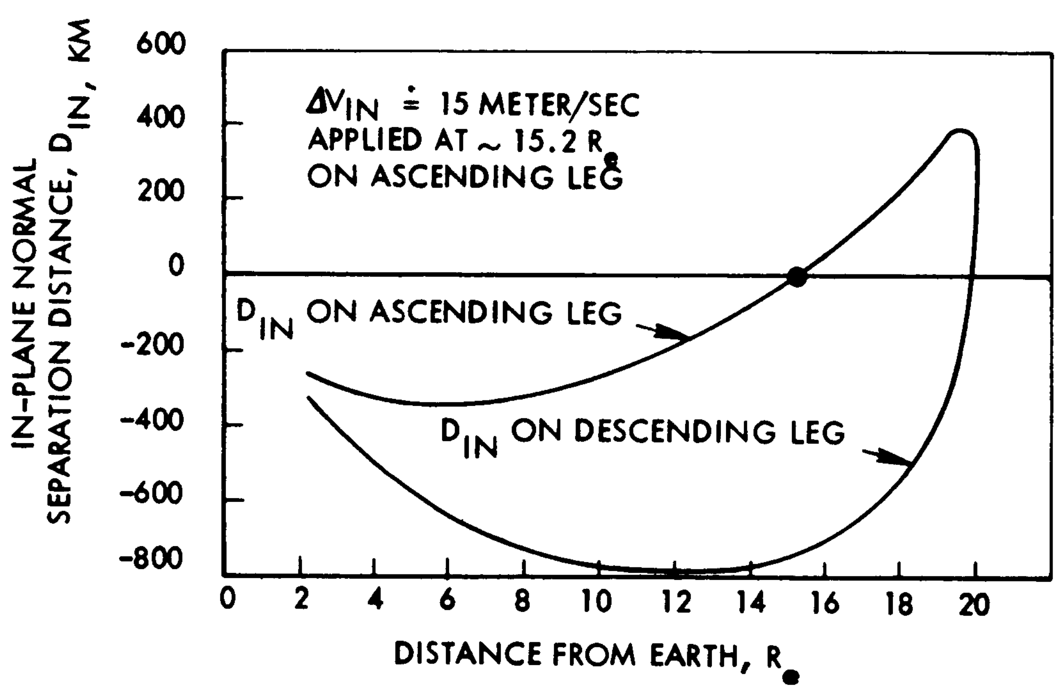


Figure 13. Typical Pattern of In-Plane Normal Separation Distance

Table 5

OUT-OF-PLANE DEPLOYMENT SENSITIVITIES

Point of ΔV_O in Application Reference Orbit		Perturbation due to ΔV_O = 1 meter/sec			
			Out of Plane Separation Distance, D_O		
				At Semi-Minor Axes	
Time from Perigee hrs	Anomaly deg	Perturbed Orbit, Δi, 10^{-3} deg	At Apogee km	Ascending Leg km	Descending Leg km
0	0	5.37	0.	2.73	-2.73
0.5	94.80	11.04	24.46	11.12	12.06
1.0	118.63	17.95	35.04	12.24	20.98
1.5	129.63	23.99	41.08	11.69	27.24
2.0	136.39	29.34	45.00	10.54	32.12
3	144.74	38.59	49.54	7.46	39.49
4	150.04	46.47	51.61	4.00	44.91
5	153.88	53.36	52.25	0.42	49.10
6	156.88	59.47	51.94	-3.17	52.40
7	159.34	64.94	50.95	-6.73	55.02
8	161.44	69.86	49.45	-10.22	57.09
9	163.28	74.31	47.55	-13.63	58.70
10	164.91	78.34	45.34	-16.94	59.92
11	166.40	81.97	42.87	-20.17	60.80
12	167.76	85.26	40.19	-23.29	61.38
13	169.03	88.21	37.33	-26.31	61.69
14	170.22	90.86	34.32	-29.23	61.76
15	171.35	93.21	31.19	-32.04	61.60
16	172.42	95.28	27.96	-34.74	61.24
17	173.45	97.09	24.63	-37.34	60.68
18	174.44	98.64	21.24	-39.82	59.95
19	175.41	99.94	17.78	-42.19	59.04
20	176.36	100.99	14.27	-44.45	57.97
21	177.28	101.81	10.73	-46.59	56.75
22	178.20	102.39	7.17	-48.61	55.39
23	179.11	102.73	3.57	-50.50	53.88
24	180.01	102.85	-0.03	-52.27	52.24
25	180.91	102.73	-3.63	-53.91	50.47
26	181.82	102.38	-7.22	-55.41	48.57
27	182.73	101.80	-10.79	-56.78	46.55
28	183.66	100.98	-14.33	-57.99	44.41
29	184.60	99.18	-17.84	-59.06	42.15
30	185.57	98.62	-21.30	-59.96	39.78

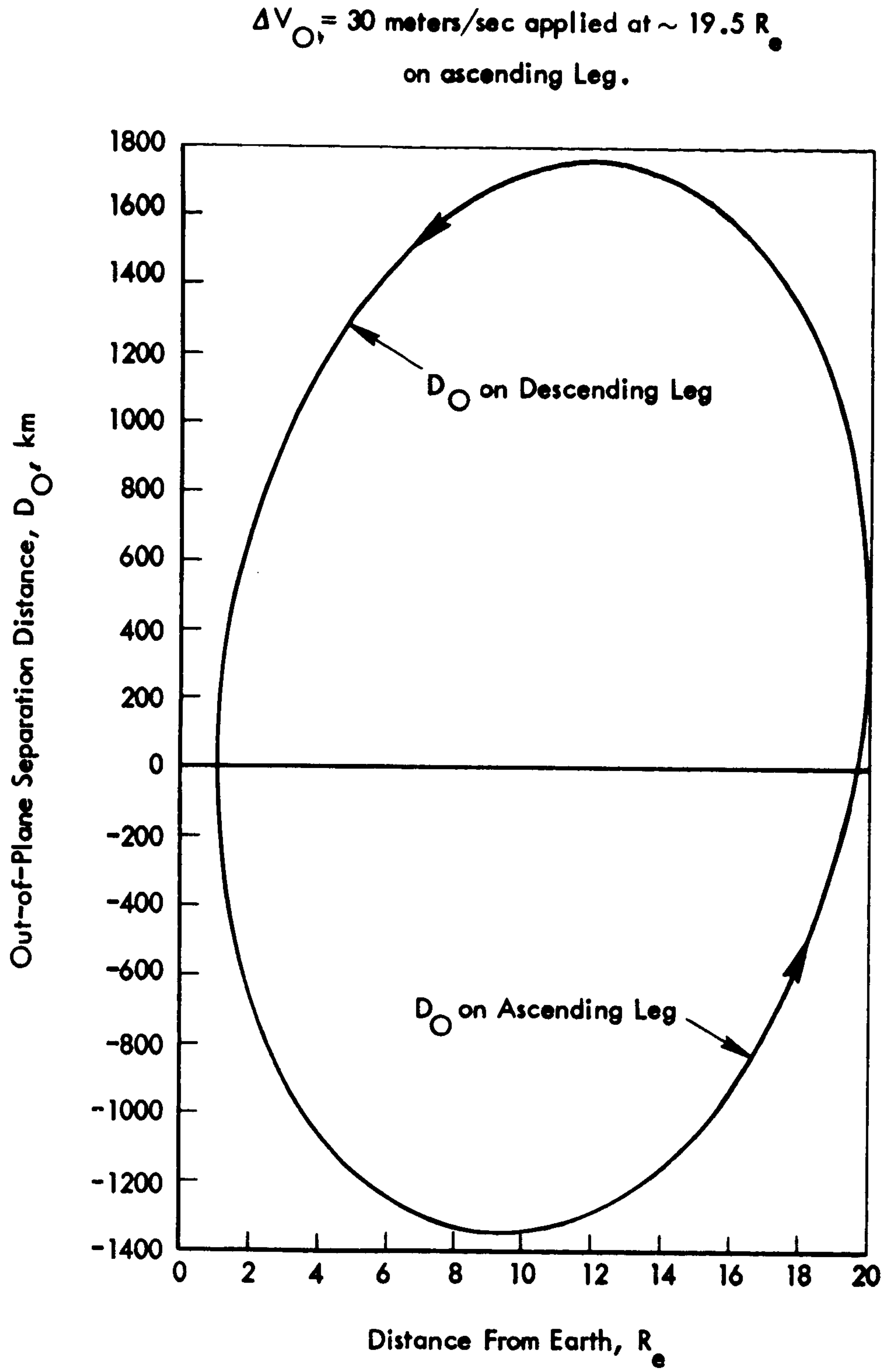


Figure 14. Typical Pattern of Out-Of-Plane Separation Distance

When the separation distances between the four satellites are resolved in any orthogonal coordinate frame, it is necessary that all components be present to achieve a non-coplanar array. Hence, it is necessary (though not sufficient) that the deployment produce all three components of the separation distance resolved in the local reference frame, i.e., D_T, D_{IN}, and D_O. To achieve the scientific objectives it is desirable that the minimum component of separation distance in any direction be in the order of 500 to 1000 km. A primary problem in deriving an acceptable deployment scheme is to achieve in-plane normal, D_{IN}, and out-of-plane, D_O, separation distances of the desired magnitude, while avoiding excessive growth in the tangential separation distance component, D_T. Since sizeable ΔV_{IN} and ΔV_O velocity increment components will be required while ΔV_T must be kept small, it is apparent that accuracy will be necessary in the control of the direction of the deployment velocity increments.

As indicated in Figure 15, the sensitivities of D_{IN} and D_O to the velocity increments attain maximum values for deployment at relatively large distances, while the sensitivity of the growth in D_T decreases steadily as apogee is approached. Hence, the magnitude of the velocity increment and the degree of directional accuracy decreases when the velocity increments are applied at relatively large distances from Earth[12].

3.5 DEPLOYMENT SCHEME

The deployment scheme that has been devised for pallet design proceeds as follows[13]:

The pallet carrying the four satellites is injected into orbit by the booster vehicle spinning at a rate of about 140 rpm with its spin axis close to the direction of the velocity vector at perigee. Using an ACS system similar to that employed by Pioneer VI, the spin axis is reoriented to become aligned with the normal to the orbit. This is accomplished by (1) processing the spin axis so that the spin axis becomes normal to the sunline and, then, (2) precessing the spin axis about the sunline to align the spin axis with the normal

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[12] See Appendix X for a description of deployment velocity and accuracy requirement.

[13] See Appendix III for a preliminary evaluation of separation distances and accuracies.

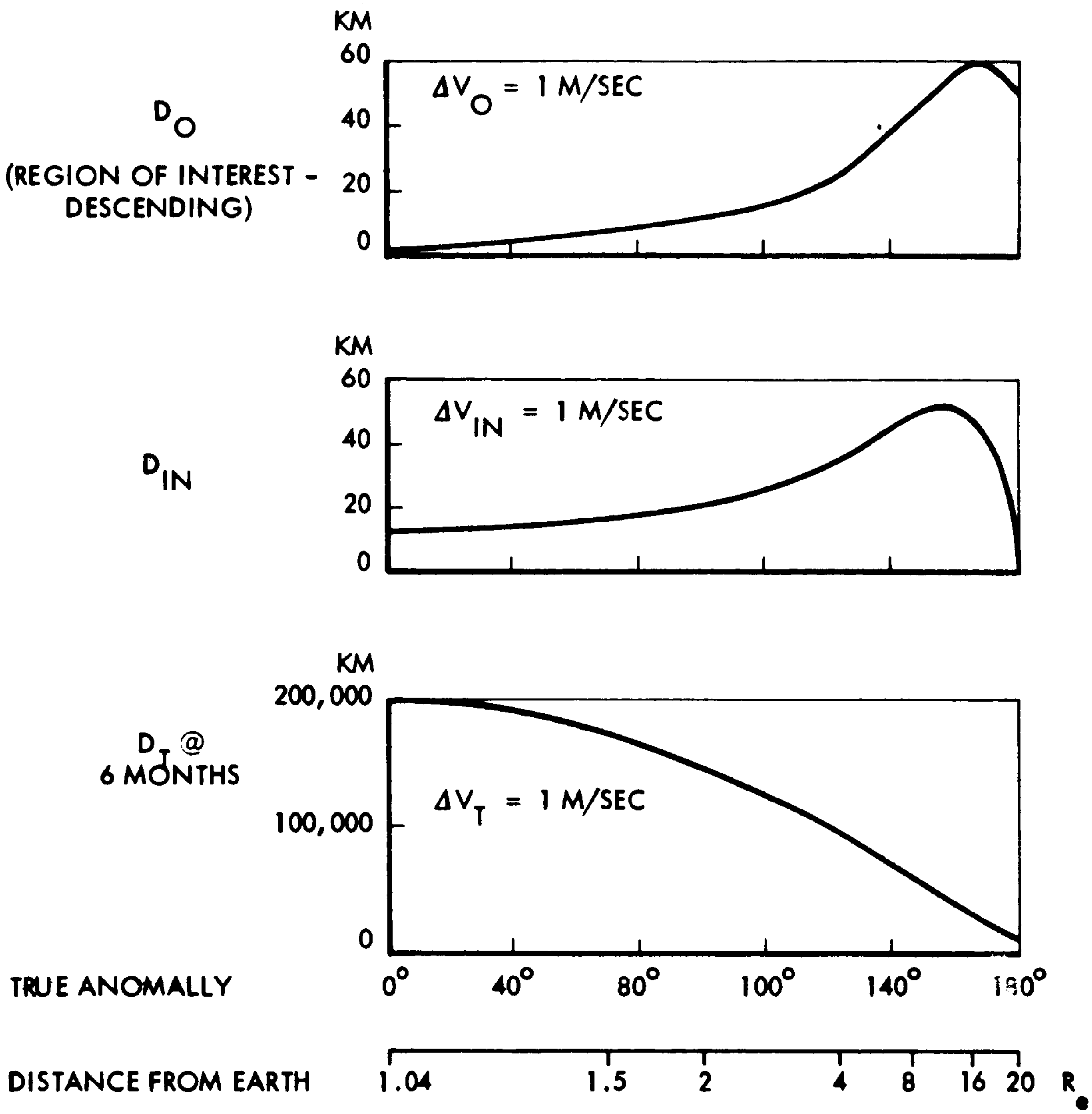

Figure 15. Variation of Deployment Sensitivities With Point of ΔV Application

to the orbit. As in the case of the Pioneer VI system, the first step is referenced to ACS sun sensors carried by the pallet. The second step is based upon aspect data obtained from infrared (IR) horizon-crossing indicators[14]. These sensors are carried by the satellites. Their data is obtained through the use of the satellites' downlink telecommunications system.

The satellites are mounted on the pallet in pairs, as illustrated in Figure 16. After the direction of the pallet's spin axis is confirmed by monitoring the IR aspect sensor data, satellite pairs 1-2 and 3-4 are separated from the pallet by spin-off. This separation is armed by ground command and is triggered by a signal from a sun sensor which is positioned so that in the nominal case the velocity increments imparted to the satellite pairs are in the in-plane direction normal to the velocity vector. In the nominal case the spin-off separation command is given when the pallet is about 15.2 R_e from Earth on the ascending leg of the orbit. This corresponds to about 10 hours from perigee passage. Correction for any known off-nominal conditions, such as off-nominal angle between the orbit frame and the sunline or an off-nominal spin rate, is made by adjustment of the point in the orbit when the command for the spin-off separation is given.

For a lateral distance between the pallet's spin axis and the center of mass of a satellite-pair of about one-half meter, and the pallet spinning at 140 rpm, the satellite pairs are separated with a velocity increment of about 7.5 meter/sec. Since the directions of the velocity increments for the pairs are opposite, the differential velocity increment is about 15 meter/sec. Hence, the distribution of the in-plane normal separation distances between satellites 1-2 and satellites 3-4 corresponds to that shown in Figure 13.

Because of the limitations on the magnitude of the spin-off velocity increment, the nominal point for the spin-off separation was selected to obtain near maximum magnitude for the in-plane normal separation distance. The selected point was biased slightly off the maximum so as to reduce sensitivity of growth in tangential separation due to directional errors.

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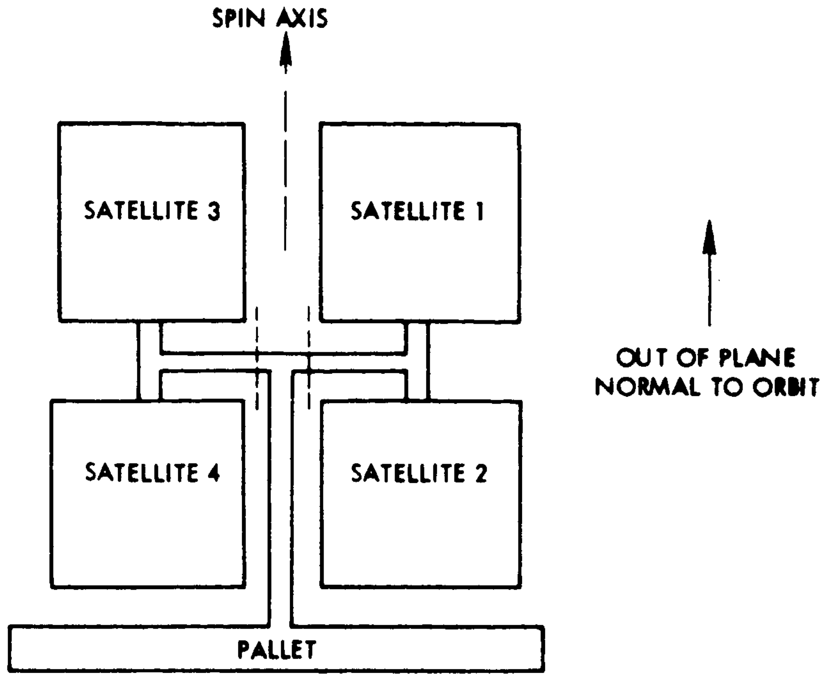
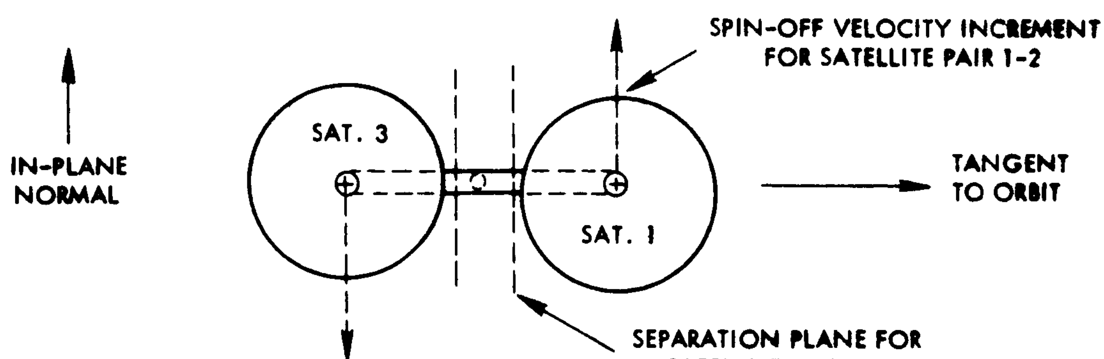
[14] See Appendix XI for a comparison of alternate aspect sensors.



a. SIDE VIEW OF MOUNTING OF SATELLITES ON PALLET



b. VIEW FROM ABOVE

Figure 16. General Arrangement of Mounting of Satellites on Pallet

Since the moment of inertia ratio for the satellite pairs is not favorable, the satellites are separated from their partners very shortly after spin-off separation. This is accomplished by axial spring separation which imparts small out-of-plane velocity increments to the satellites.

Inasmuch as it is not feasible to obtain an out-of-plane velocity increment of the magnitude required by means of spring separation, small solid rockets, thrusting along the satellites' spin axes, are used. An upward out-of-plane velocity increment of 15 meter/sec is imparted to satellite 1 and a downward velocity increment of 15 meter/sec is imparted to satellite 2 by firing their solid rockets on ground command. This command is transmitted when the satellites have reached the point on the ascending leg of their orbits where the tangent to the orbit is perpendicular to the sunline. In the nominal case this will occur at about 20 hours from perigee passage when the satellites are about 19.5 R_e from the center of the earth. This point is selected for the out-of-plane deployment so that any errors that cause the spin axes to be tilted away from the normal in the direction of the sunline will not contribute to the generation of a tangential velocity increment component. It is noted that the orientation of the satellites' spin axes is inherited from the pallet. With the Pioneer VI type pallet ACS, the angle between the spin axis and the sunline is fixed in advance and cannot be adjusted in flight to correct for an off-nominal orbit orientation. On the other hand, the rotation of the pallet's spin axis about the sunline can be adjusted in flight. With the choice of the point for out-of-plane deployment, this adjustment is sufficient to permit correction for off-nominal orbital conditions to be made.

The differential out-of-plane velocity increment between satellites 1 and 2 of 30 meter/sec produces the distribution of out-of-plane separation distances shown in Figure 14. Out-of-plane velocity increments are similarly applied to satellites 3 and 4, by command-firing of their axial solid rockets. However, since out-of-plane separation has been obtained between satellites 1 and 2, out-of-plane separation between satellites 3 and 4 is not absolutely essential. Their solid rockets are used: (1) to keep all satellites identical in design; (2) to avoid orbital period differences between satellites 3-4 and satellites 1-2 arising from second order effects of velocity increments perpendicular to the velocity vector; and (3) to assure that some tangential separation between satellites 3 and 4 will be obtained. Tangential separation between the partners of one of the satellite's pairs is required to obtain a non-coplanar array. This tangential separation may be obtained inadvertently as a result of deployment errors. If necessary, the orientation spin axis could be deliberately tilted slightly and the point of out-of-plane deployment chosen to assure the tangential separation of satellites 3 and 4.

The errors made in the deployment have been studied in some detail. The results, which are summarized in Tables 6 to 8, were based upon conservative estimates. They indicate that the restriction on the growth of tangential separation distance can be met.

3.6 DEPLOYMENT SIMULATION AND ARRAY HISTORY

The deployment concept devised for the mission was based upon guidelines developed from simplified representations of the effects of the deployment velocity increments and the orbital perturbations. In order to verify the concept, a digital computer simulation of the deployment and subsequent array history was run.

For this purpose a nominal pallet orbit of 750 km x 20 R_e was used with a due-east launch from ETR at the time of day chosen to minimize inclination to the ecliptic and on the Julian date appropriate for a lead time of 35 days before passage of the sunline over apogee. The satellites were deployed in the following manner:

a. At 10 hours after perigee passage, (about 15 R_e on the ascending leg of the orbit), the spin-off separation from the pallet was simulated by applying an inward in-plane velocity increment of 7.3 m/sec to satellite pair 1-2 and an outward in-plane normal velocity increment of 7.5 meter/sec to satellite pair 3-4. It was assumed that the spin-off separation produced no reaction on the pallet.

b. At 20 hours after perigee passage (about 19.5 R_e on the ascending leg of the orbit), the firing of the axial solid rockets of satellites 1 and 2 was simulated by applying an upward out-of-plane velocity increment of 15 m/sec to satellite 1 and a 15 m/sec downward out-of-plane velocity increment to satellite 2. It was assumed that the spin axes of the satellites was tilted by about 1.5 degree from the normal to the orbit in the

Table 6

ESTIMATES OF SPIN-OFF VELOCITY INCREMENT DIRECTIONAL ERRORS

Error Sources	Magnitude of Source Error	Contribution to Directional Error, deg	Contribution to Tangential Velocity, meters/sec	Contribution to Period Difference, % of Period
1. Uncertainty in reaction forces at separation	.05 lb-sec	.2	.05	.004
2. Sun sensor error	.1 deg	.1	.03	.002
3. Uncertainty in action time	.14 ms	1.2	.31	.024
4. Residual error in adjusting for off-nominal conditions	.1 deg	.1	.03	.002
RSS		1.25	.32	.025

Table 7

ESTIMATES OF OUT-OF-PLANE VELOCITY INCREMENT DIRECTIONAL ERRORS

Error Source	Contribution to Directional Error, deg	Contribution to Tangential Velocity, meters/sec	Contribution to Period Difference, % of Period
1. Tilt of pallet's spin axis relative to normal to orbit			
Residual error in adjusting for off-nominal conditions	.1	.05	.002
Altitude sensor error	1.0	.52	.018
2. Shift in satellite's spin axis relative to pallet's spin axis			
Disturbance on separation from pallet	.01	-	-
Disturbance on separation from partner	.04	.02	.001
Coning motion	.1	.05	.002
3. Deviation in direction of velocity increment relative to satellite's spin axis (due to misalignments)	.1	.05	.002
RSS	1.0	.52	.018

Table 8

SUMMARY OF ESTIMATES IN GROWTH OF TANGENTIAL SEPARATION DISTANCE

Cause of Tangential Separation Distance Growth	Growth in Tangential Separation Distance, km After 6 months in Orbit				
	At Perigee	At 5 R_e	At 10 R_e	At 15 R_e	At 20 R_e
1. Errors incurred in application of spin-off velocity increment	39,100	16,000	9,400	5,700	2,040
2. Errors incurred in application of out-of-plane velocity increment	28,100	11,500	6,750	4,100	1,500
3. Difference in perigee altitude	15,600	6,390	3,750	2,300	800
RSS	50,000	20,400	12,000	7,200	2,600

NOTES:

1. The spin-off error is due almost entirely to uncertainty in action time of separation mechanism. A simple system having a 1.4 millisecond uncertainty was assumed. With a more complicated system this error could be reduced by an order of magnitude.

2. The out-of-plane velocity increment error is due almost entirely to the attitude sensor error which was assumed to be 1 degree.

3. It is assumed that a perigee altitude of about 370 km will be obtained for the lowest satellite.

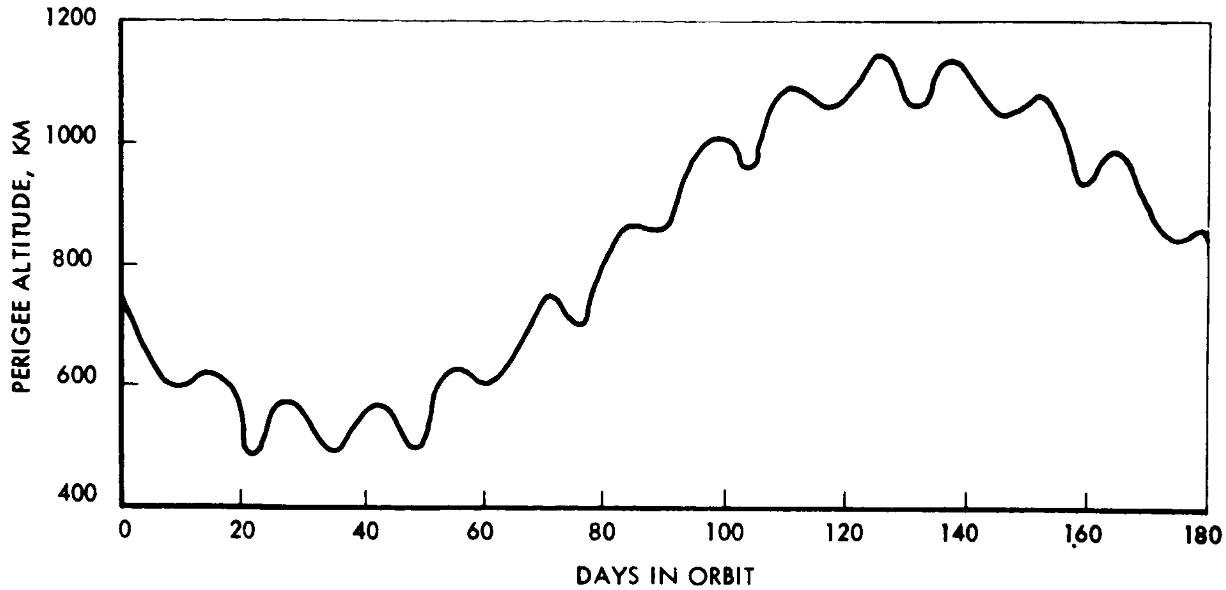
direction of the sunline, which at this point in the orbit is perpendicular to the orbit. Hence, the out-of-plane velocity increments were accompanied by small in-plane-normal velocity increments: -.39 m/sec for satellite 1; +.39 m/sec for satellite 2.

c. At 32 hours after perigee passage (about 18.5 R_e on the descending leg of the orbit), the firing of the axial rockets of satellites 3 and 4 was simulated by applying a 15 m/sec upward out-of-plane velocity increment to satellite 3, and a downward 15 m/sec velocity increment to satellite 4. At this point the tangent to the orbit is nearly parallel to the sunline. Hence, as a result of the tilt of the spin axes, these out-of-plane velocity increments are accompanied by small tangential velocity increments: -.39 m/sec for satellite 3; +.39 m/sec for satellite 4.

The simplified analysis of the array characteristics indicated that tangential separation distance between the partners of one satellite-pair would be required to obtain a non-coplanar array. Therefore, the foregoing out-of-plane deployment of satellites 3 and 4 was deliberately designed to obtain growth of the tangential separation distance between these satellites. Based upon the approximate deployment sensitivity calculations, a growth of tangential separation distance between satellites 3 and 4 in six months, in the order of 11,000 km near 10 R_e was expected.

The history of the position coordinates and the velocity component of the four satellites and the pallet during the six months following deployment was computed using Space-General's 712 Trajectory Program. This program determines position and velocity using a step-by-step integration process which takes account of all gravitational effects such as solar and lunar attraction and earth oblateness. Also included were the effects of atmospheric drag.

The trajectory history data were processed to obtain the histories of the orbit parameters and the intersatellite separation distances. Figures 17 and 18 show the histories of the most pertinent parameters of the pallet's orbit. No unexpected variations appeared in the histories of perigee altitude, inclination to the ecliptic, or true anomaly of the common line. Oscillations in orbit period of the magnitude shown in Figure 18 had not been anticipated. These oscillations are apparently due to second order effects of the solar and lunar perturbations. Fortunately, the satellites' periods followed very similar



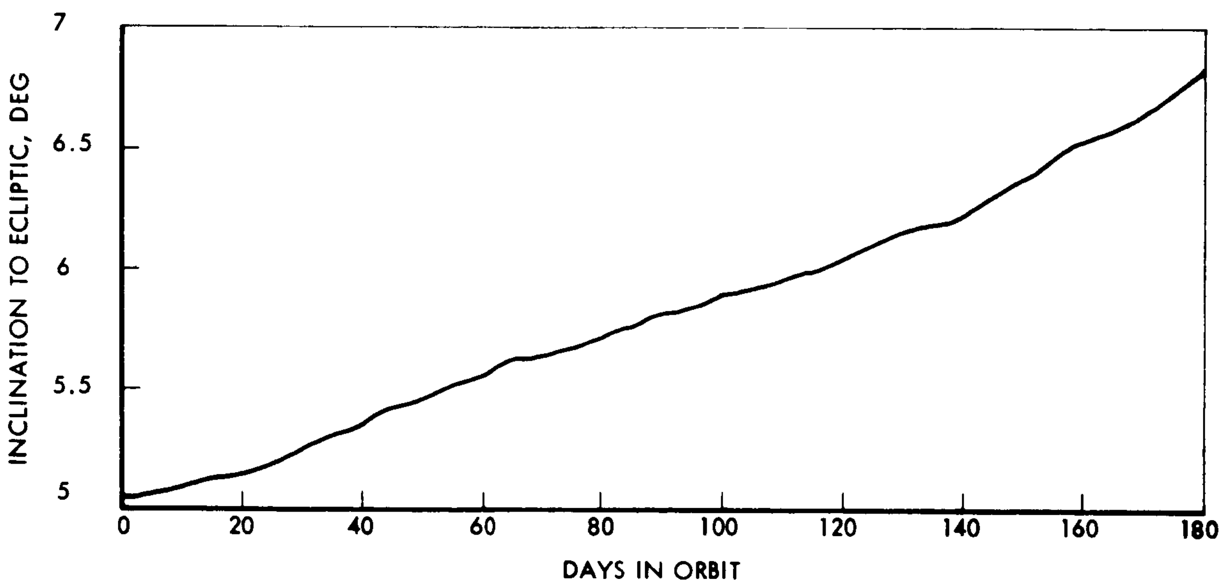


Figure 17. History of Perigee Altitude and Inclination for Pallet's Orbit

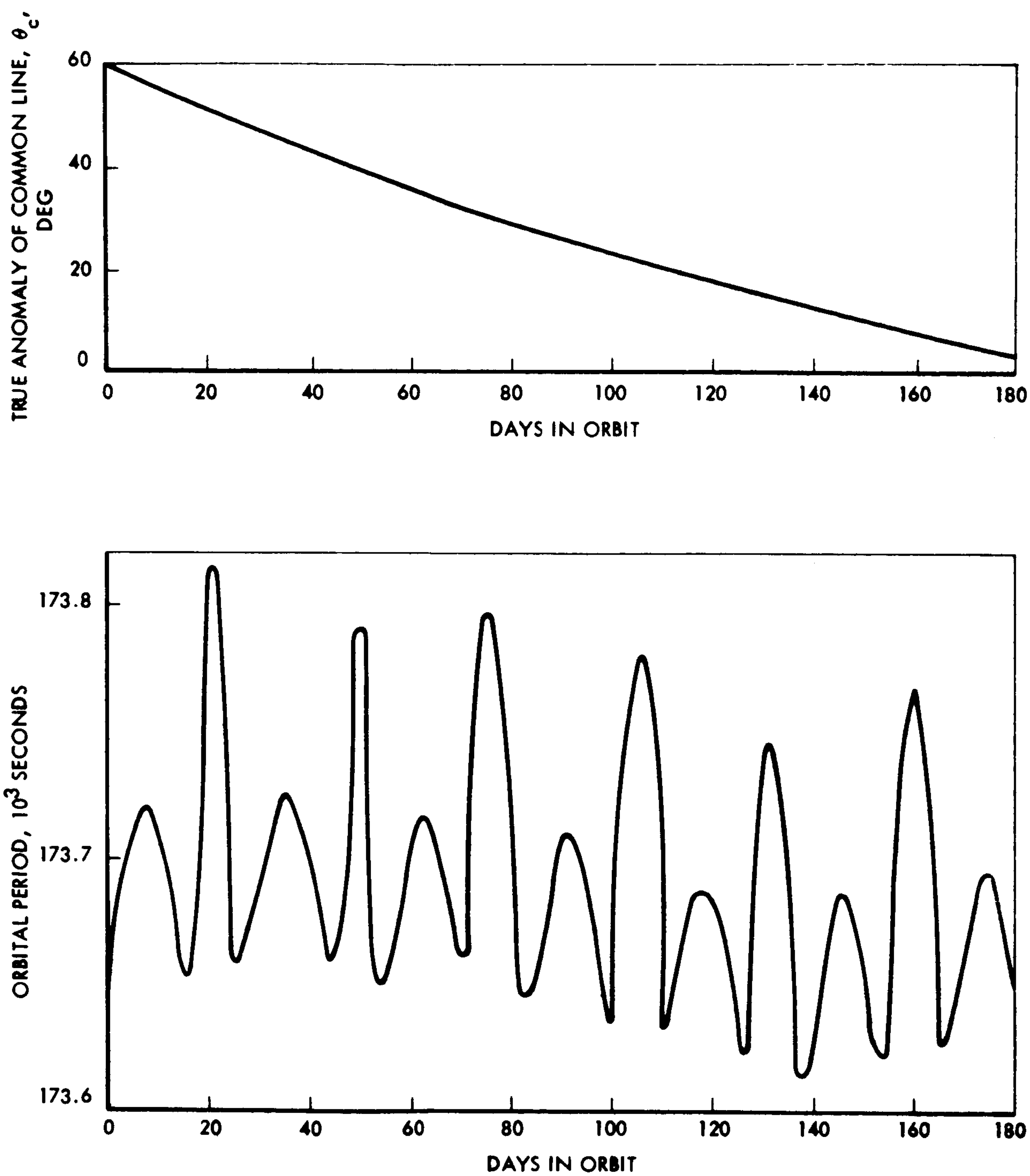


Figure 18. History of θ_c and Orbital Period for Pallet

oscillations so that this effect did not produce significant changes in the growth of tangential separation distances, which did agree closely with the expected order of magnitude.

For the purpose of examining the characteristics of the array, the position of each satellite was referenced to the pallet's position. Thus, the pallet's position coordinates were first subtracted from the position coordinates of each satellite. The separation distances between each satellite and the pallet were then resolved into tangential, in-plane normal and out-of-plane components and plotted as illustrated in Figure 19 so as to assist three-dimensional visualization of the array.

In order to check on the non-coplanarity of the array, the distance of each satellite from the plane passing through the other three satellites was computed. The smallest of these distances gives the smallest dimension of the array measured in any direction. Figures 20 and 21 show the distributions obtained for these separation distances at the various times during the first 6 months in orbit.

The early history of the array of the intersatellite distances followed the expected course. At the outset, the array is nearly co-planar; on the ascending leg this is due mainly to the absence of separation in the in-plane normal direction; on the descending leg, although the in-plane normal separation distances are ample, the line between satellites 1 and 2 and the line between satellites 3 and 4 are both nearly vertical so the four satellites lie close to a vertical plane. As the tangential separation distance between satellites 3 and 4 grows, the non-coplanarity of the array on the descending leg improves. This continues to about the 60th day in orbit. At about this time the effects of differential orbital perturbations start to show up. These effects tend to degrade non-coplanarity on the descending leg and improve non-coplanarity on the ascending leg.

In spite of an awareness of the possibility of differential perturbation effects, the particular nature of the results obtained had not been foreseen. These results were traced to differential earth oblateness perturbations which arise in consequence of the differences in inclination resulting from the out-of-plane deployment velocity increments. This causes relative

Satellite Array at 14.53 R_e on Descending Leg of Orbit on 15th Day

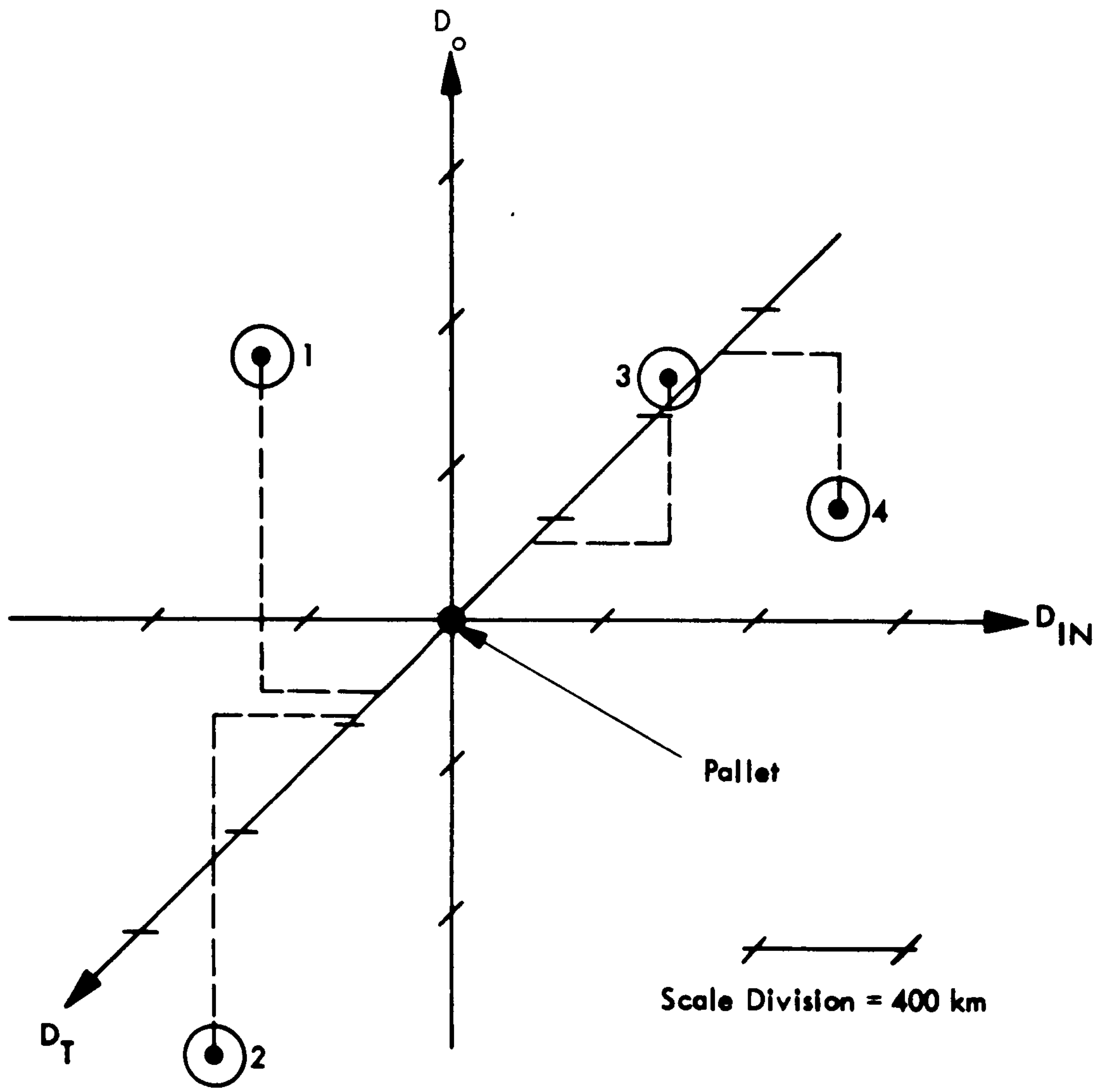


Figure 19. Typical Array of Intersatellite Separation Distances

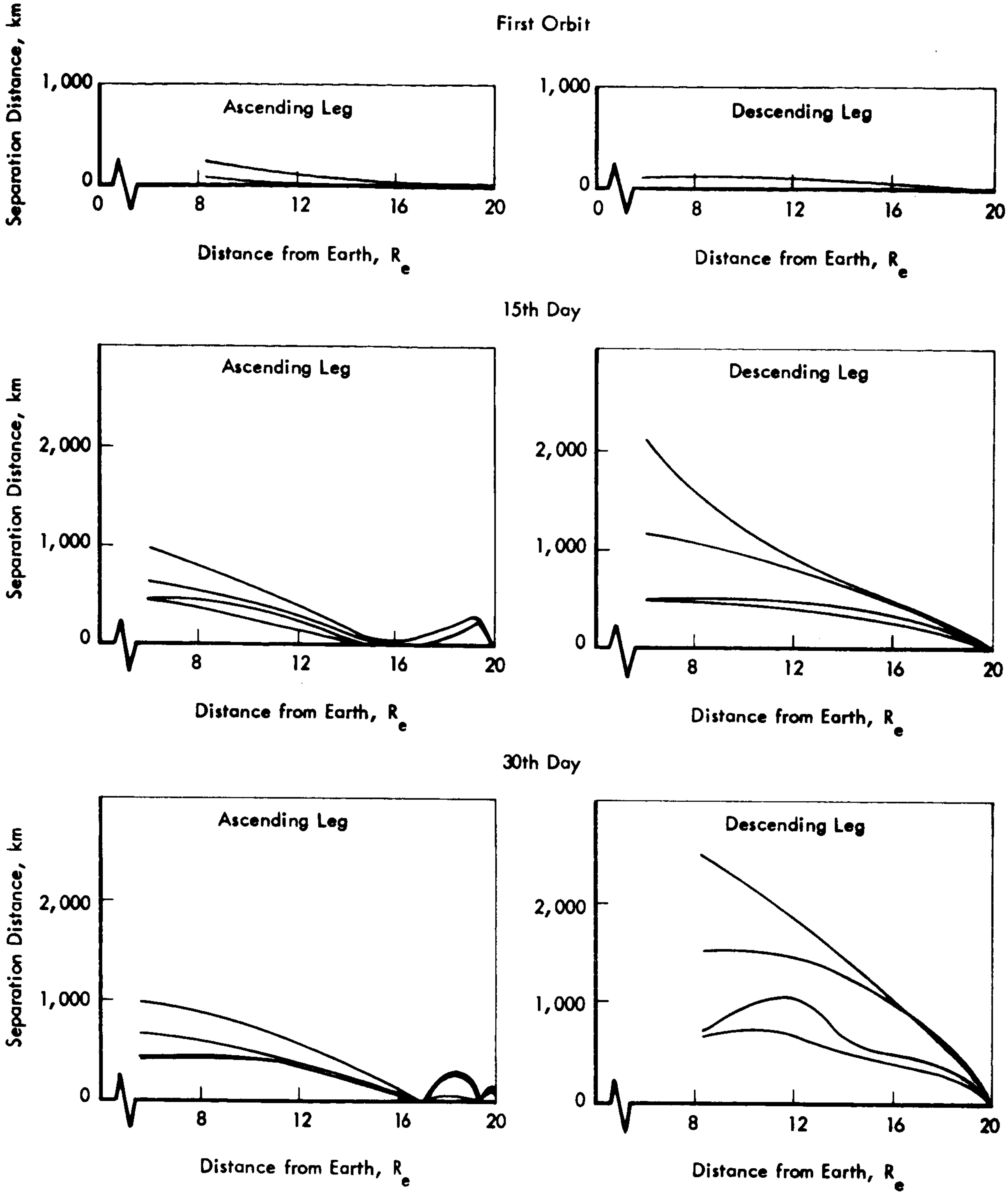


Figure 20. Separation Distances for 1st, 15th and 30th Days (Distances Shown Are From Each Satellite to The Plane of the Other Three)

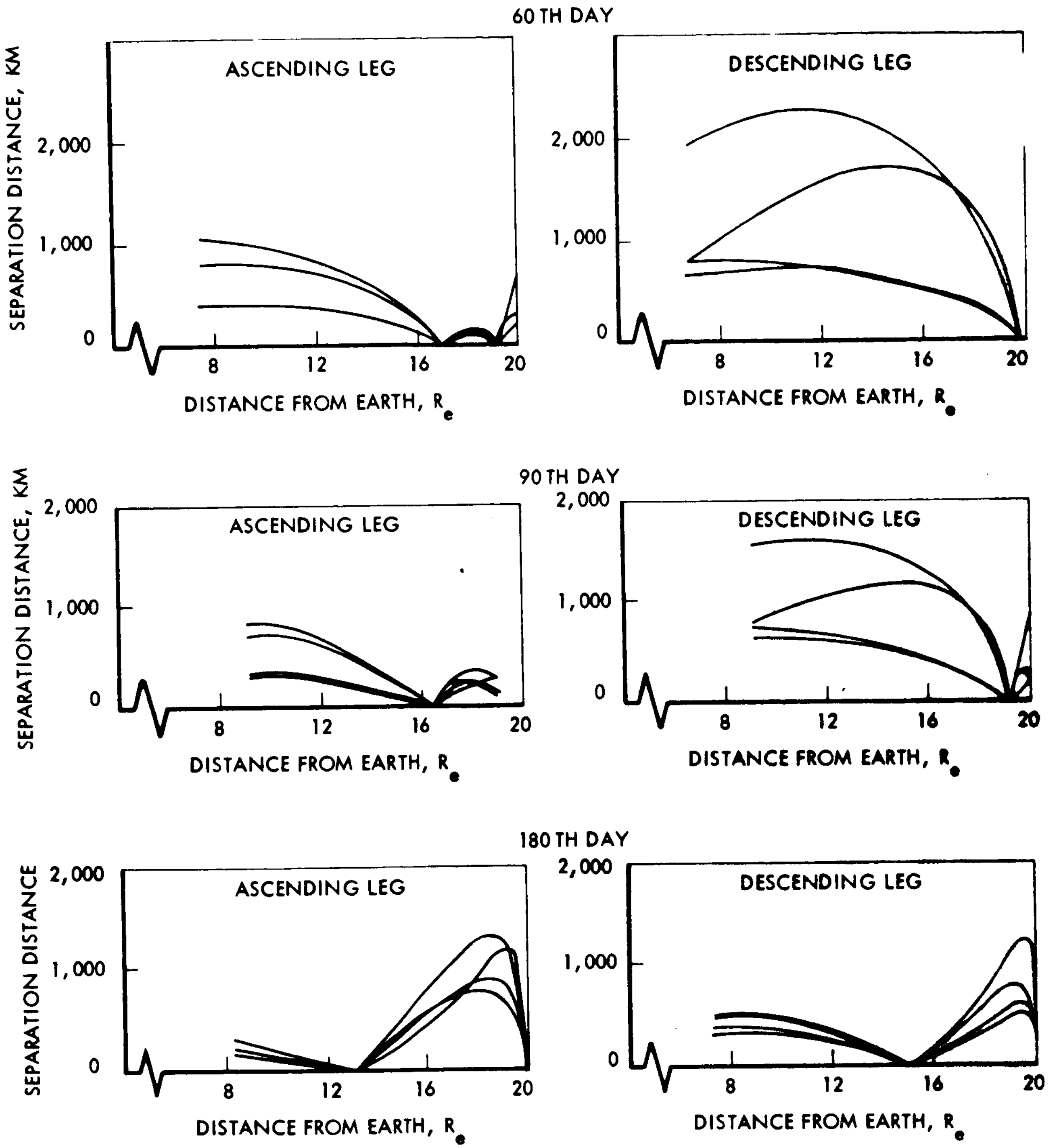


Figure 21. Separation Distances for 60th, 90th, and 180th Days (Distances Shown Are From Each Satellite to the Plane of the Other Three)

rotations of the line of apsides, i.e., a twisting of one satellite orbit relative to another, that changes the in-plane normal separation distances. In principle, this effect can be estimated by use of amplified techniques for evaluating the orbital perturbations. This is shown by the comparison of results given in Table 9.

Since the nature of the significant differential orbital perturbations effects is now known, it is possible that the deployment scheme could be revised to take advantage of the effect, or at least avoid undesirable consequences.

3.7 ATTITUDE REORIENTATION

A spin-axis, precession control ACS, similar to that used by Pioneer VI, provides the simplest, most reliable and lightweight implementation for execution of the pallet's reorientation maneuver. This system also avoids any requirement for despinning and respinning the pallet. It is comprised of:

a. A single-solenoid-valve, single-nozzle cold gas pneumatic system

b. Four sun sensors and control electronics

c. A passive precession damper

The sun sensors are simple photodetectors shaded to obtain the desired view fields, as illustrated in Figure 22. Each sensor produces a discrete output signal when the sunline is within its view field. The valve is held open during the interval that a sensor signal is present in accordance with the command-selected operational mode.

The reorientation maneuver is made in two steps. In the first step, the spin axis of the pallet is precessed until it is nominally perpendicular to the sunline. This is accomplished by commanding the jet to go on when the sunline enters the view fields of either sensor A or B. In the case of the multiple satellite mission, the initial angle between the spin axis and the sunline will range from 35 to 75 degrees. Since the sunline is initially in the forward hemisphere about the spin axis, the first maneuver will always start with the forward looking sensor, i.e., sensor A, operating the solenoid valve. The

Table 9

DIFFERENTIAL PERTURBATION OF ORBIT PERTURBATION

	Pallet	Satellite 1	Satellite 2	Satellite 3	Satellite 4
Longitude of Ascending Node (Relative to Ecliptic Plane)					
Initial Value	0	-12.9	+16.4	-10.7	14.8 deg
Final Value* from Trajectory Program	-47.1	-50.5	-39.6	-50.4	-14.7 deg
Final Value using Simplified Estimate of Earth Oblateness Perturbation	-46.6	-50.1	-38.6	-50.0	-44.3 deg
Inclination (Relative to Ecliptic Plane)					
Initial Value	5.06	5.84	4.60	6.01	4.35 deg
Final Value from Trajectory Program	6.90	7.99	5.71	8.30	8.22 deg
Final Value using Simplified Estimate of Earth Oblateness Perturbation	6.78	7.97	5.43	8.22	5.55 deg
True Anomaly of Line Common to Orbital and Ecliptic Planes					
Initial Value	60.0	47.0	76.2	49.5	74.9 deg
Final Value from Trajectory Program	4.5	1.4	11.8	1.3	6.5 deg

*Final values apply to 180 days in orbit.

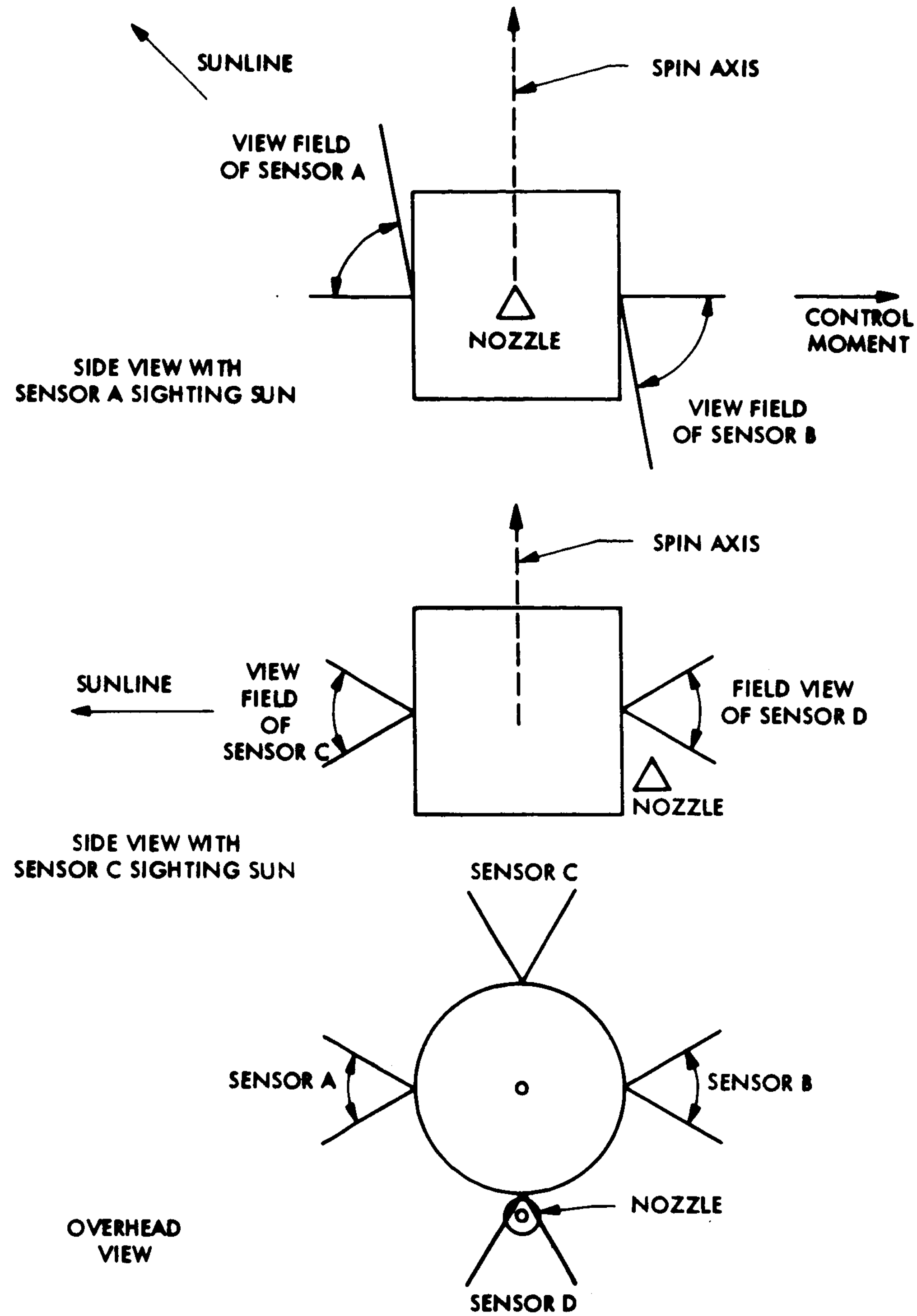


Figure 22. Arrangement of ACS Sun Sensors

mounting position of the nozzle on the pallet and the phasing of the jet on-time obtained from sensor A are such as to generate a moment about the axis that lies in the spin-axis/sunline plane; it is perpendicular to the spin axis, and is directed away from the sunline. This causes the spin axis to precess away from the sunline about the axis that is mutually perpendicular to the spin axis and the sunline. The precession is stopped and the first step of the reorientation maneuver is completed when the spin axis has moved far enough so that the sunline comes into the view field of sensor B; i.e., the view field of sensors A and B are slightly overlapped and the presence of signals from both sensors during a spin cycle is used to stop the first step of the maneuver.

Nominally, the angle between the spin axis and the sunline will then be 90 degrees. However, the plane of the overlap of the sensors' view fields can be adjusted to obtain a specified angle between the spin axis and the sunline, within a wide range. This adjustment can be used to accommodate the angle between the normal to the orbit and the sunline which, in general, will differ somewhat from 90 degrees because of the inclination of the orbit to the ecliptic plane.

In the second step of the maneuver, the spin axis is precessed about the sunline to align it normal to the orbit. This is accomplished by commanding the jet to go on when the sunline is in the field of view of sensor C. The mounting position of the nozzle and the phasing of the jet on-time produced by sensor C are such as to generate a moment in the direction mutually perpendicular to the spin axis and the sunline. This results in a counterclockwise precession of the spin axis about the sunline bringing the spin axis normal to the orbit after a rotation of about 90 degrees.

The second step of the maneuver is stopped by ground command when data obtained from IR horizon-crossing indicators carried by the satellites indicate that the desired orientation has been achieved. In the event of an overshoot, the jet is commanded to go on when the sunline enters the view field of sensor D. This results in a clockwise rotation about the sunline. During the second step of the maneuver a small error in the angle between the spin axis and the sunline might be introduced. This error can be remove by returning the system to the first step mode. By repeated use of the available modes, the orientation error can be trimmed out until the residual error is primarily that due to the aspect sensors.

The maximum rotation of the spin axis required for the first step is about 55 degrees. The maximum rotation for the second step is about 100 degrees. Hence, the total maximum rotation is about 155 degrees. For a moment of inertia of 32 slug-ft^2, a spin rate of 140 rpm, and a moment arm of about 2 feet; an impulse of about 630 lb-sec is required for this rotation. An 11-pound N_2 gas supply, with a specific impulse of 70 seconds, will provide an impulse of 770 lb-sec leaving 140 lb-sec as a margin for inefficiencies, such as correcting for overshoot, etc.

Since more time can be made available for the reorientation maneuver than was the case for Pioneer VI, it will probably be advisable to reduce the step size to about 0.1 degree to allow the achievement of higher final orientation accuracy. This may be obtained with a thrust level of about 2.7 pounds.

Reduction of the step size also reduces the demands on the precession (wobble) damper. If necessary, the requirements on the precession damper can be further reduced by inhibiting the opening of the valve so that a specified number of spin cycles elapses between cycles during which time the valve is not permitted to open. This can be accomplished with the use of simple countdown circuitry.

To obtain greater precision, the use of a viscous ring precision damper may be desirable. This would permit a very small residual cone angle to be obtained. Interference with the booster is avoided by keeping the damper caged prior to separation from the booster, and then uncaging it by firing an explosive valve.

ACS systems similar to that recommended for the multiple satellite have been investigated in some detail in earlier SGC spacecraft design work. These efforts have included analog computer simulations and preliminary design studies. The results of this work, plus the obvious success of the Pioneer VI system, leave little doubt regarding conceptual feasibility of a system of this type.

3.8 SPIN AXIS DRIFT

As a result of the pallet's reorientation maneuver, the satellites will inherit a spin axis orientation very closely aligned to the normal to the orbit. Maintenance of this orientation is desired to keep the view fields of the experiment sensors sweeping across the ecliptic plane, and is required for effective use of a fan beam downlink transmission antenna. To maintain good communications coverage, the drift of the direction of the spin axis due to disturbance torques must be small relative to the half-width of the fan beam which is later established as about 12 degrees.

Detailed computations were made to determine the drift in the spin axis orientation[15]. These computations took account of:

a. Aerodynamic torques

b. Solar pressure torques

c. Magnetic torques

d. Gravity gradient torques

Only the first two were found to have significant effects during the first six months in orbit. The drifts due to the aerodynamic torque and the solar pressure torques are proportional to the axial separation between the centers of pressure and the center of mass. For the current reference satellite design, this separation was unusually large because of the low station of the center of mass (~3.5 inches below the geometric center of the satellites). This low center of mass station resulted from the design approach used to obtain the required inertial characteristics for the pallet assembly. However, in spite of the adverse location of the center of mass, the spin axis drifts were not unduly large at the perigee altitudes of interest. In addition, the direction of the aerodynamic torque tended to be opposite to the direction of solar torque, so that to some extent a benefit was obtained from a partial cancellation of effects. As shown in Figure 23 for the perigee altitude of interest, i.e., above 350 km, the drift of the spin axis due to the combined effects of the disturbance torques remains below 2 degrees.

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[15] See Appendix XII.

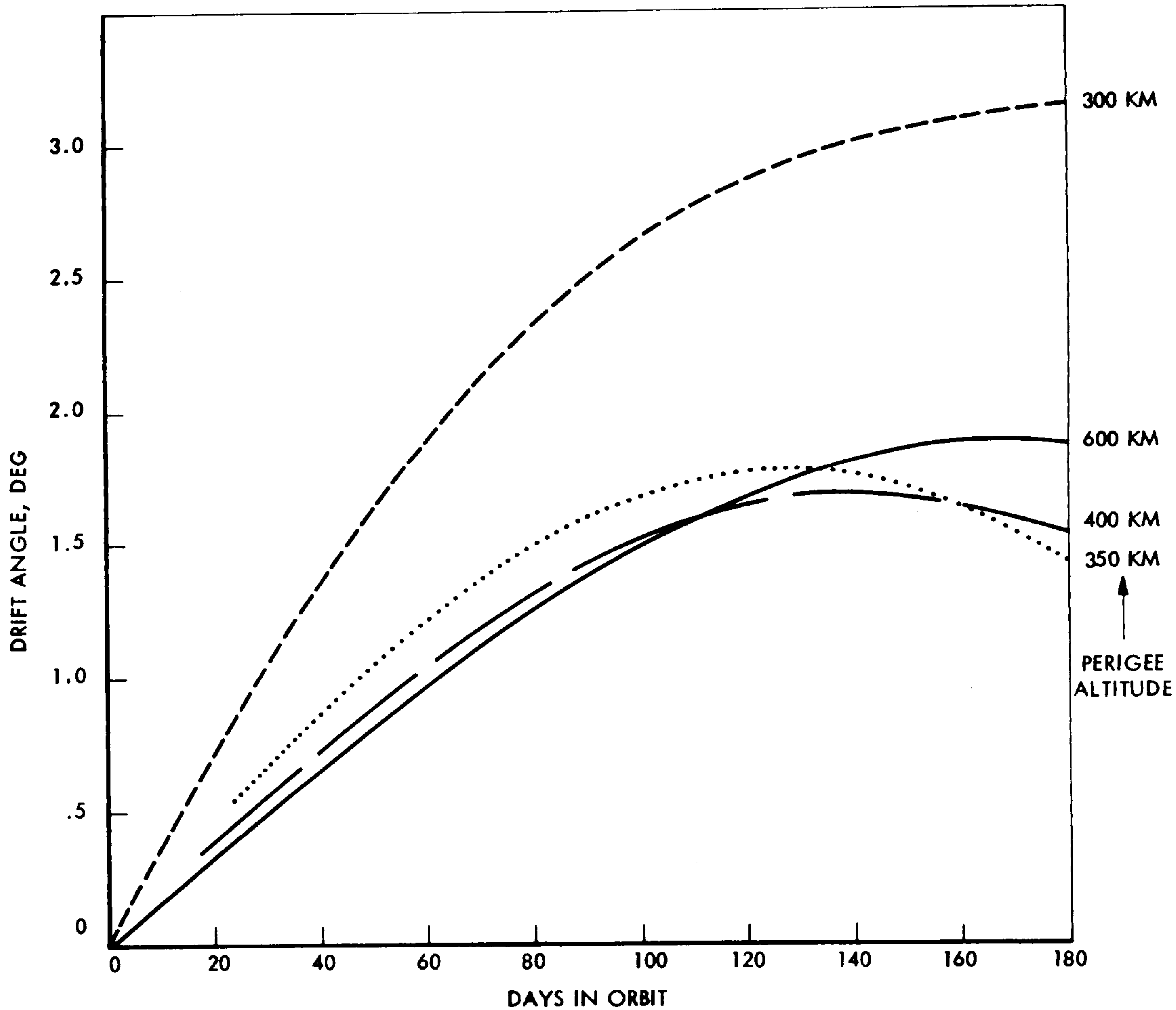

Figure 23. Combined Effect of Torques On Spin-Axis Drift Perigee Altitudes - 300 To 600 km

Section 4

COMMUNICATIONS AND DATA HANDLING

The material covered in this section consists of a description of the communication system from the output of the instruments, through the data processing equipment, to the transmitter and antenna. In addition, a description of required data storage is included. Material describing the command system, including the command antenna, receiver and decoder, is presented. A description of the Goddard Range and Range-Rate system as it applies to the Multiple Satellite Program is provided, with specific reference to intersatellite range measurement accuracies. The STADAN station operating characteristics and STADAN station coverage as a function of orbital position are shown to be very important in the system design.

In general, the communication/data handling system requirements can be met with state-of-the-art techniques using hardware which is currently available. Certain improvements in subsystem operation can be anticipated over and above that described in this section, assuming normal developments between this time and the time at which it will be necessary to commit to specific hardware. Areas which are expected to benefit from such improvements in technology are identified, but none of the performance parameters described herein are contingent upon such developments.

The communications system has been designed with a primary goal of providing the necessary support for the scientific instrument complement. To maximize the return of scientific data, two different techniques of data processing will be described:

a. Use of a basic, fixed-format telemetry sequencer

b. Use of a reprogrammable data processor to provide capability for changing spacecraft operations after the system is in orbit.

4.1 COMMUNICATION REQUIREMENTS

The communication subsystem must support, primarily, the operation of a set of scientific instruments and, secondly, aspect and engineering sensors related to orientation and performance measurements for each satellite and the pallet. The pallet communications subsystem is a simple one consisting of an antenna, a command receiver and decoder, and a tracking beacon. This system will be used to control orientation of the spacecraft system prior to deployment and also to initiate the deployment sequence. A tracking beacon at S-band will be used to track the spacecraft during the boost phase, through injection, and separation from the upper stage vehicle.

The satellite system is primarily optimized to support the scientific instrument complement. To this end, the information rate to support the scientific instruments was maximized such that this rate, as a fraction of the total downlink telemetry rate, is as large as possible. Factors which influence the percentage of total data rate to be used for the scientific instruments include:

a. The necessity for utilizing a parity bit for every word

b. The need for status monitoring of satellite subsystems

c. The necessity for command verification

d. The need for continuous aspect monitoring

The nominal downlink data transmission rate is 1280 bits per second, of which 1050 bits are used for scientific data. Thus, in the Multiple Satellite Program, a substantially greater fraction of available communications channel capability has been provided for the scientific instruments than exists, typically, in other scientific satellite programs.

The communication system must work into the NASA STADAN network. Consideration has also been given to the use of the MSFN, should this network become available during the time period in which the multiple satellite system will be operational. At the suggestion of Ames Research Center there are three principal STADAN stations to be used for reception: Tananarive, Santiago, and Orroral. A basic requirement is to minimize the use of the 85-foot antennas; hence, all telemetry link calculations have been performed assuming the use of only the 30-foot antennas in the STADAN (and MSFN) systems.

To track the individual satellites and to obtain data relative to range and intersatellite separations, use will be made of the Goddard Range and Range-Rate (GRARR) system at S-band. This system requires the use of a separate transponder in each satellite. Characteristics and requirements for this system are described in a following paragraph.

4.2 STADAN INTERFACE[16]

As noted above, consideration will be given, primarily, to using the 30-foot antennas only in the STADAN network. It is assumed that the characteristics of the antennas and operating equipment of the MSFN are equivalent to that of the STADAN network. Each STADAN station will have at least four receivers capable of operation through a multicoupler from one antenna, to demodulate the signal from each of the four satellites simultaneously (provided that the orbital characteristics are such that the four satellites can be viewed simultaneously by one ground antenna). Each of the STADAN stations will have full capability to perform range and range-rate measurements via the GRARR system. The GRARR system will have its own independent antennas, receivers, and computing equipment for determining satellite range and range-rate. The STADAN network will have a ground data link with a capacity of at least 2400 bits per second to the Goddard Space Flight Center (NASCOM). It is anticipated that, if a requirement exists for real time decommutation at any one of the STADAN stations, such decommutation and reformatting will be performed via the automatic decommutation equipment available at each station. Such data will be reformatted by the computer at each station and transmitted to GSFC via the 2400 bits-per-second data lines for additional processing and display.

The antennas to be used at each of the STADAN sites are 30-foot parabolas with a gain of approximately 44 dB and a noise figure of approximately 200°K. These antennas have uncooled parametric amplifiers and down-converters located in the feed-horn assembly for reception at S-band. The converter will down-convert the received S-band carrier frequency to the 136-138 MHz band. This signal is then sent to the multiple receivers.

The STADAN network will have the capability for utilizing several different command systems. These command systems are: (1) a tone-only

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[16] See Appendix XIII

system, (2) a tone-digital system, and (3) a PCM system. While all command links are currently 148 MHz, S-band command links will be in use for the Multiple Satellite Program. The command structure which will be required for the operation of each of the satellites is of the order of 80 different commands. It is apparent that the PCM standard or some variation thereof will be required. If modifications are made to this system by GSFC prior to committing to detail design, such modifications and improvements will be incorporated in the Multiple Satellite Program.

4.3 SATELLITE TRACKING

To perform the ranging function as noted above, the GRARR system is used. A transponder is provided within each spacecraft for obtaining range and range-rate measurements with the necessary degree of accuracy to insure that scientific mission requirements are met. The transponder is defined in GSFC document No. S-531-P-17, together with the associated ground station characteristics. It is anticipated that this transponder will be built to appropriate GSFC specifications.

The ground antennas used are 30-foot paraboloids with a nominal gain figure of 42 dB. Each antenna has a down converter of the uncooled paramp type with a band width of 100 MHz. Each station has four receivers which receive nominally at 136 MHz, or 2250 MHz. Various modes are available for operation which include auto-track, ranging, telemetry functions, etc. Various tracking rates are available for the expected doppler shift, in loop bandwidths ranging from 10 Hz to 3000 Hz and tracking rates ranging from 200 Hz/sec^2 to 20,000,000 Hz/sec^2. The satellite transponder anticipated for use on the Multiple Satellite Program has a nominal weight of seven pounds, a power consumption of seven watts and a transmit power of one-half-watt.

Because the satellites, separately and when on the pallet, are spinning, undue errors caused by doppler shift may be introduced reducing the performance of the ranging system. An analysis was performed to evaluate the magnitude of this doppler shift. The case analyzed concerned the spinning pallet, since the doppler shift would be considerably worse than for an individual satellite. The pallet spin rate could be as high as 3 rps and the distance from the antenna to the spin axis could be as much as 3 feet. The analysis was performed considering a simplified spinning system. It is assumed that the time required to make a single measurement is small compared to the spin rate and the rate of change of the nominal range[17]. If the ranging frequency is about 2000 MHz, the maximum doppler shift is 125.6 Hz. This doppler is well within the tracking rate of the smallest loop bandwidth of the receiver as described in GSFC document No. F-531-P-17, Section 4.5.4.2.3. Additional doppler effects which must be considered are the changes in translation doppler caused by vehicle rotation, or phase acceleration. For a frequency of 2000 MHz, the phase acceleration is 2400 Hz/sec^2. Again, this is well within the capability of the receiver, but might require the use of a larger loop bandwidth, thus resulting in a drop of 10 dB in the acquisition and tracking threshold.

The basic single point accuracies for the GRARR system are (1) range error less than 15 meters rms, (2) range-rate error less than 0.1 meter/sec, and (3) direction error less than 0.1 degree. These accuracies use the 100 kc ranging tone and a ground transmit frequency of 1800 MHz (nominal) with a transmitter power of 10 kw. The received frequency from the satellite in a nominal 2253 MHz. Based upon a simplified but conservative analysis,[18] the ephemeris accuracy achieved should allow the prediction of absolute position for each satellite to better than 3 km and the intersatellite range to better than 4 km. These values assume an operational sequence involving some 40 rangings per orbit on initial orbit to establish the ephemeris, followed by 3 rangings per orbit from each of 2 stations to maintain the ephemeris. For the actual case, the use of smoothing techniques in the special orbit prediction programs should allow even greater accuracy for these ranging frequencies.

4.4 GROUND STATION AVAILABILITY

To control the duty cycle requirement on the STADAN stations (as requested by Goddard STADAN personnel and by NASA OTDA) to a maximum of six hours per station per day, analyses were performed to define the

[17]See Appendix XIV for detailed analysis.
[18]See Appendix XV for details.

time-visibility of the satellites at the stations. Three sets of computations were performed:[19]

a. Visibility computations designed primarily to guide the selection of ranges for the tape recorder playback

b. Computations designed to establish the real-time coverage that will be available

c. Computations designed to verify the recorder readout coverage based upon the characteristics of the communications system design

Additionally, as will be described later, communications link analyses were performed assuming a set of down-link real-time data rates, and non-real time data rates (for tape recorder playback) equal to 24 times the real-time rate.

With playback at 30,720 BPS, one hour per satellite is required for recorder readout. The communications link analysis indicates that with a nominal one-watt transmitter power, playback will be restricted to range values below approximately seven and one-half-earth radii. The analysis of view angles available from the stations, plus the coverage of an appropriate satellite fan-beam antenna pattern, indicates that coverage will be available down to and in some cases below 4 R_e. Therefore, the region during which tape recorded data can be played back is conservatively defined at between four and seven and one-half-earth radii on both the ascending and descending legs. The analysis further indicates that under worst case conditions, taking into account both ascending and descending legs, somewhat more than four station-hours per orbit are available to view the satellite system. Under the maximum satellite separation distance conditions, each satellite comes into view at intervals of about one hour and no two satellites fall within the beamwidth of one receiving dish, thus, four station-hours per orbit would be required to receive the data from all satellites under worst conditions. The computer results indicate that this condition is met in all cases; indeed, useable view times are typically more than two hours at a given station location,

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[19] See Appendix XVI for details.

and that two or more stations are typically in view during the readout interval. Table 10 is a sample of the computer printout which provides the source data for Figure 24 and which verifies the adequacy of station availability.

For ranges in excess of seven and one-half earth radii there will be at least one, and more often two stations available which can perform deployment, tracking, ranging and communication functions over the entire orbit. The inadequacy of a system based on real-time transmission only for data collection arises from two points:

a. Frequently, after deployment, only one satellite may fall into the beamwidth of a station's 30-foot antenna dish. Thus, coverage by up to four dishes in parallel would be necessary on a continuous basis to support the mission. This coverage is generally unavailable, even with no duty cycle restriction.

b. The potential requirement for four antenna coverage in parallel results in a total of 96 dish-hours per day. This level of support would require essentially a 100 percent duty cycle by four single-dish stations, which greatly exceeds the six hours/day/station maximum.

Thus, the real-time communications capability can be utilized only during deployment and for near-apogee coverage early in the system life (while the separations are small). Fortunately, this is also the period during which the real-time capability is the most desirable.

4.5 DATA ACQUISITION AND DOWN-LINK CONSIDERATIONS

The communications and data subsystem has as its primary purpose the acquisition and transmission to Earth of data from the scientific instruments and, secondarily, to provide sufficient measurements to monitor the engineering status of the spacecraft and spacecraft subsystems. A functional block diagram illustrating the major components of this subsystem is included in the satellite block diagram, Figure 32 of Section 5.1.

It is assumed that two basic signal types are to be processed: (1) digital signals from the scientific instruments, and (2) analog signals from the spacecraft engineering sensors. These signals are multiplexed through two

Table 10

SAMPLE PRINTOUT OF VERIFICATION OF RECORDER PLAYBACK COVERAGE

Apogee Radius = 20 RE

KEY

Initial Conditions

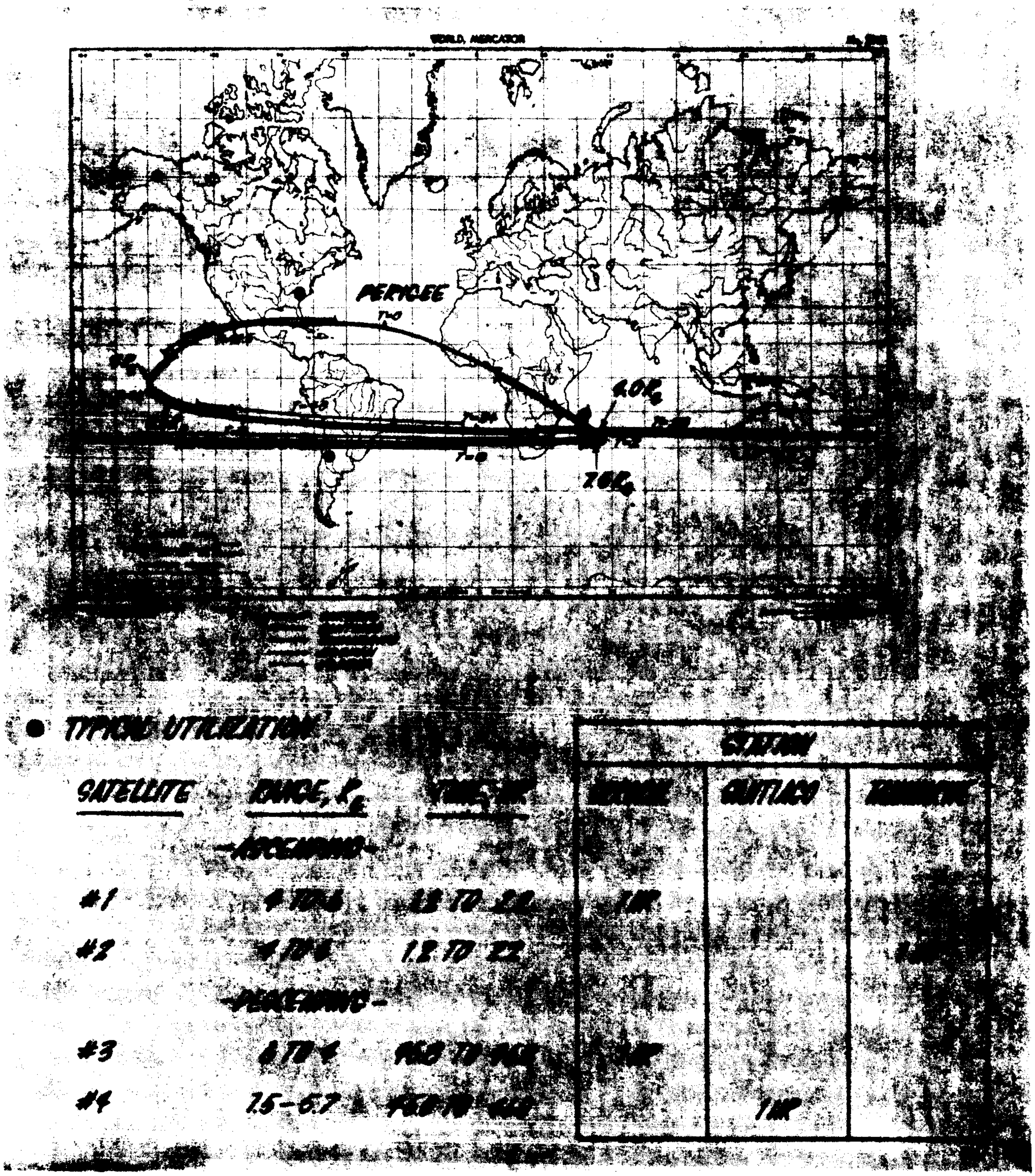
Satellite	Apogee Alt / ECC	Perigee Alt / SMA	Arg Per / Period	Node / TP	T Anomaly / VA	Incl / VP
1	121052.50	280.00	150.0000	-165.0000	0.	28.5000
	0.90078402	67037.43	172737.061	-0.	0.557104	10.673019

Receiving Station Location

Number	Latitude	Longitude	Radius	Name
1	-35.63	148.95	6371.95	Orroral
2	35.20	-80.00	6371.88	Rosman
3	-33.13	-70.68	6371.69	Santiago
4	-17.00	48.00	6371.00	Tananarive
5	64.98	-147.50	6371.29	Ulaska

COVERAGE

Station	Time On	Radius On	Time Off	Radius Off	DT
Orbit No. 1					
1.	0.64	2.54	2.87	7.29	2.23
2.	0.	0.	0.	0.	0.
3.	0.	0.	0.	0.	0.
4.	0.19	1.29	2.87	7.29	2.69
5.	0.	0.	0.	0.	0.
Orbit No. 2					
1.	48.62	2.54	50.86	7.29	2.23
2.	45.42	6.75	47.53	2.00	2.11
3.	44.83	7.75	47.36	2.50	2.53
4.	48.17	1.29	50.86	7.29	2.69
5.	45.56	6.50	47.27	2.75	1.70
Orbit No. 3					
1.	96.61	2.54	98.84	7.29	2.23
2.	93.41	6.75	95.60	1.75	2.20
3.	92.97	7.50	95.25	2.75	2.28
4.	96.15	1.29	98.84	7.29	2.69
5.	93.55	6.50	95.25	2.75	1.70
Orbit No. 4					
1.	144.59	2.54	146.82	7.29	2.23
2.	141.39	6.75	143.59	1.75	2.20
3.	140.95	7.50	143.14	3.00	2.19
4.	144.14	1.29	146.82	7.29	2.69
5.	141.53	6.50	143.23	2.75	1.70



SGC/1046

Figure 24. Station Availability

submultiplexers and a main multiplexer. The main multiplexer is of the variable word-length type, i. e., the resultant telemetry format is composed of words whose length is determined by the accuracy or operating requirements of the instrument and sensors. In this manner, specialized circuitry in each instrument is eliminated with a slight increase in the circuitry required in the sequencer and main multiplexer. The net effect is to reduce, to a significant degree, the total circuitry in each satellite system. The decommutation procedures and equipment available in the STADAN stations are compatible with such a format. If real-time decommutation is not required, such formats are decommutated easily by computer at GSFC.

The sequencer will provide repetitive control pulses or coded words as required by the instruments or other support subsystems of each satellite. Provision will be made to accept commands from the ground stations to modify or inhibit such control functions, if operational or scientific reasons require such operational modifications. The command subsystem provides a suitable receiver and associated decoder to provide a means for the control of the operation of the satellite from a STADAN station. It is assumed that, because of the nominal number of command words, required as shown in Table 11, a digital word command system similar to that described in GSFC document No. X-560-63-2 will be used. This system is easily implemented since existing STADAN control consoles are available for use. Commands received from the ground will be used, primarily, to turn on and off the tape recorder, transmitter, ranging transponder, and individual instruments. The bulk of the available commands will be associated with the instruments; however, it is anticipated that in normal operation these commands would be utilized less frequently than the power switching commands.

It should be noted that, while the system described above is somewhat different from most telemetry systems in that it features a variable word-length format, no format changes are permitted once the hardware elements have been designed and built. An alternative technique would be to use a variable program data processor. A flexible data processor applicable to the Multiple Satellite Program has been designed, a block diagram of which appears in Figure 25. The design objectives were:

Table 11

TYPICAL COMMAND FUNCTIONS

I. PALLET

(2) Beacon On-Off

(1) Uncage Precession Damper/Third Stage Separation Backup

(1) Enable ACS

(3) ACS Mode-Select and Initiate

(1) Stop ACS

(1) Activate Spin-Off

II. SATELLITES

(2) Transponder On-Off

(2) Transmitter On-Off

(2) Data Mode Select (R. T. or Record)

(1) Spring Release Satellite Pairs

(1) Uncage Precession Damper

(1) Fire Solid Motor

(1) Deploy Boom

(3) Record Cycle Select

(12) Instruments On-Off (6 Inst.)

(~50) Instrument Functions

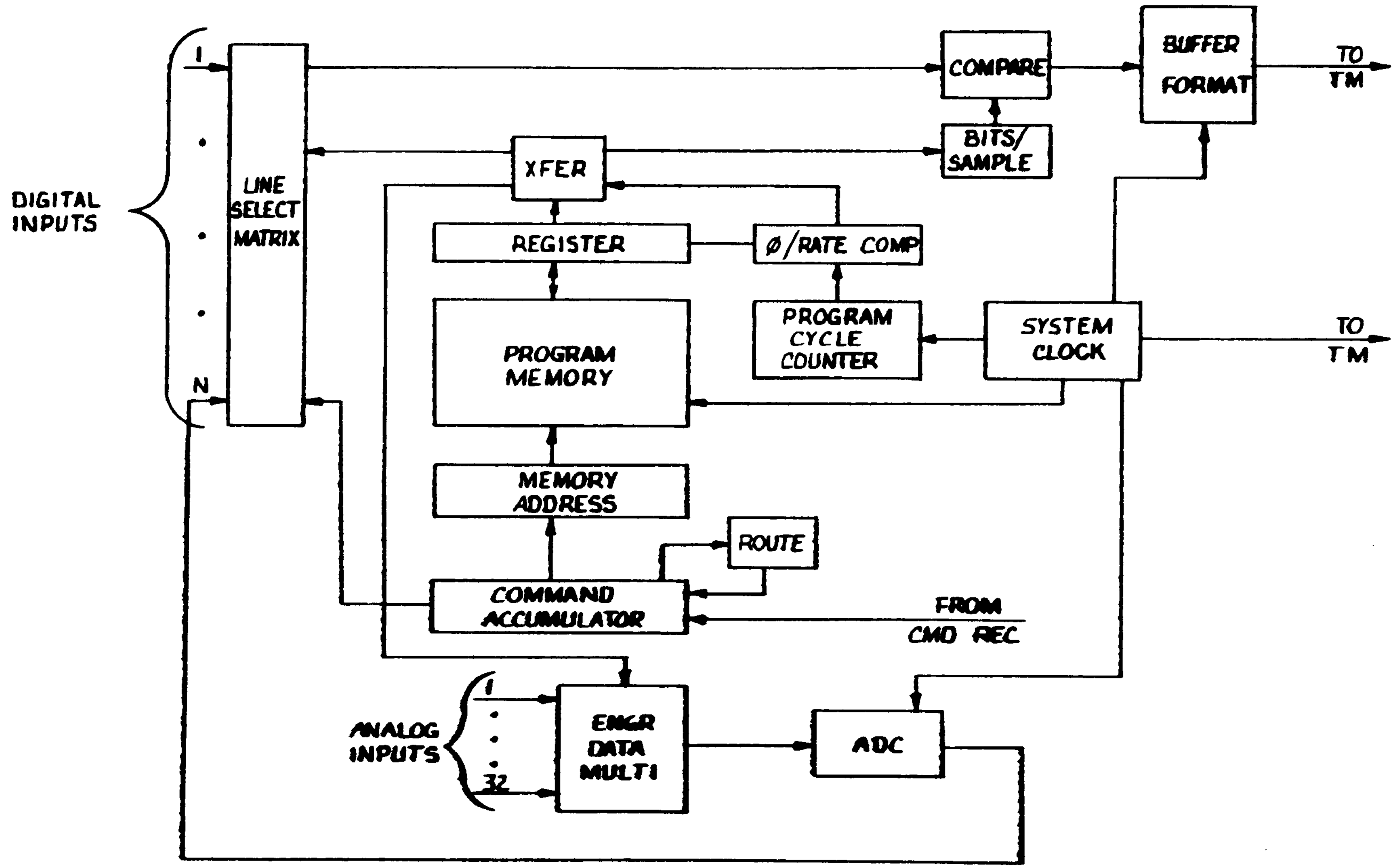


Figure 25. Data Processor, Variable Program Option

a. Minimum weight and power consumption

b. Maximum reliability

c. Strict adherence to the data requirements of the principal investigator

d. A flexible design to accommodate instruments without requiring hardware modifications

e. Variable word lengths to meet required instrument data accuracies

These objectives have been met for the utilization of:

a. An all-digital interface between any instrument and the data processor

b. A small memory (reprogrammable) which provides all control and sequence function information

The interface used is illustrated in Figure 26. Each interface function transmits to the instrument control and clock signals and the instrument transmits data synchronously to the processor upon receipt of control signals. In general, the instrument contains the following items: a command accumulator/decoder, an instrument engineering multiplexer, and an analog-to-digital converter. The use of this interface provides the following features:

a. All instrument data processor interfaces are identical

b. System cable weight is a minimum

c. The number of physical connections is reduced

d. Instruments may be substituted without modifying support system hardware

Reference to the block diagram shows that the data processor provides all sequencing and control functions. The processor collects data from the instruments, formats the data, and sends it to the modulator transmitter. The major functional elements of the processor are:

a. Program memory

b. Comparator/decoder address register

c. Bits-per-sample comparator

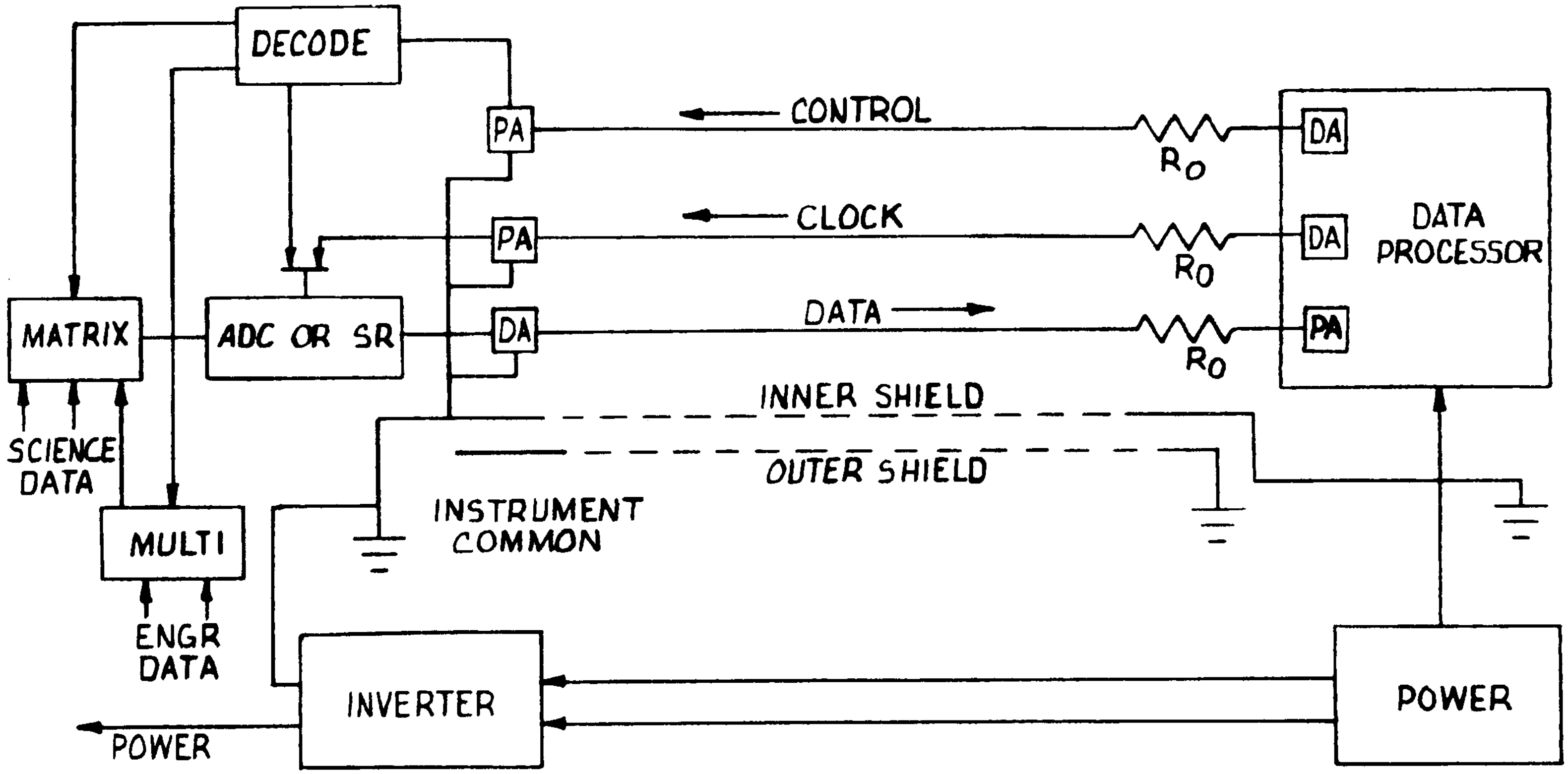


Figure 26. Instrument/Data Processor Interface, Variable Program Option

d. Output buffer

e. Line-select matrix

f. Local engineering data multiplexer/converter

In the data processor, the line-select decoder and channel-address decoder act on input signals from the address register as follows: the control line to the proper instruments is activated sending the control signal from the data processor to the instrument decoder. The clock line to the same instrument is activated sending a clock signal to the instrument which synchronizes the data output circuits. The data line is closed and serial data is accepted from the instrument. Engineering data within the instrument is treated similarly except that the sampling rate is much lower. The input signals to control the channel address decoder are generated from the program memory.

The nominal program memory contains sixty-four 28-bit words. The first syllable of the 28-bit word is a 14-bit phase-rate syllable, the second is a 9-bit channel function syllable and the third is a 5-bit number-of-data-bits-per-sample syllable. There is a unique program word in the memory for each data source to be sampled and each function to be controlled by the data processor.

An advantage obtained with this system is that the data bits/sample syllable of the control program word is used to determine the number of bits per sample to be accepted from each data source, thus permitting variable word length words commensurate with exact instrument requirements to be read out easily and to be changed in orbit if required. The data from the instruments passes through the line select decoder to the formatter. The bits per sample comparator stops the data stream when the desired number of bits have been transferred into the buffer formatter. The buffer formatter attaches frame sync signals and time tags as required and delivers output data to telemetry at a constant rate without any gaps in the format.

4.6 DOWNLINK FORMAT

A typical format has been prepared and is shown in Figure 27. Basic information relating to word lengths has been derived from supporting Ames Research Center documentation. The format illustrated consists of a main frame which is 1280 bits long and is equal to the design data storage and real-time rate. The main frame is composed of five identical subframes, each of which is denoted by unique subframe ID number. Each subframe is composed of 21 variable length words with each word corresponding to the stated accuracy of measurement for a given instrument or instrument channel. Parity bits are appended to each word to aid in the reduction of undetected word error rate. The format, as shown, is completely compatible with existing PCM decommutation procedures at the STADAN sites. If no real-time display is required all such data is processed by computer. If time code words are introduced every five subframes, the time between samples of the given instrument is offset from an equal interval between successive samples. If this should prove objectionable from a scientific standpoint, reduced length frame sync and time code words could be introduced within each subframe, thereby eliminating master frame sync and time code.

4.7 COMMUNICATIONS LINK ANALYSIS

A link analysis was performed by programming a computer to iterate the parameters in the design control table, considering range, data rate antenna gain, and transmitter power variations. Certain of these calculations, particularly those in the critical area of tape recorder playback, were plotted by machine and displayed graphically for easy reference and comparison. Shown in Table 12 is a typical downlink design control table based upon a nominal one-watt transmitter which covers two cases, (a) maximum slant range of 80,000 miles, and (b) recorder readout at approximately 7.5 R_e.

4.8 TRANSMITTER SELECTION

The selection of the required transmitter RF power output must consider the following major factors:

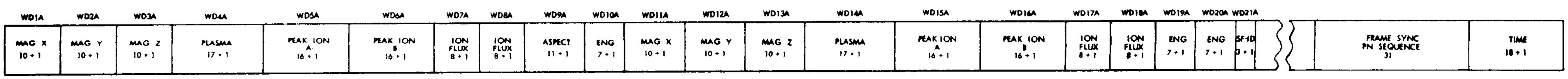


Figure 27. Multi-Satellite Typical Down-Link Telemetry Format

Table 12

DOWNLINK DESIGN CONTROL TABLE

	Parameter	Value: 1280 BPS 80,000 mi.	Value: 30,720 BPS 7.5 R_e	Tolerance
1.	Transmitter Power	+30 dBM	+30 dBM	+1.0, -0.5
2.	Transmit Circuit Loss	-1.0 dBm	-1.0 dB	$\pm$0.1
3.	Transmit Antenna Gain	+6.4 dB	+6.4 dB	+0, -0.4
4.	Transmit Antenna Pointing Loss	0 dB	0 dB	--
5.	Space Loss at 2250 MHz	-202.1 dB	-193.6 dB	--
6.	Polarization Loss	-3 dB	-3.0 dB	--
7.	Receiving Antenna Gain	+44.0 dB	+44.0 dB	$\pm$1.0, -0.5
8.	Receiving Antenna Pointing Loss	0 dB	0 dB	--
9.	Receiving Circuit Loss	-0.5 dB	-0.5 dB	$\pm$0.1
10.	Net Circuit Loss	-156.2 dB	-147.7 dB	
11.	Total Received Power	-126.2 dB	-117.7 dB	+2.2, -1.6
12.	Receiver Noise Spectral Density, N/B ($T_{system} \approx 200^oK$)	-175.7 dBm	-175.7 dBm	$\pm$0.8
13.	Carrier Modulation Loss (θ = 76°)	-12.4 dB	-12.4 dB	$\pm$0.5
14.	Received Carrier Power	-138.6 dBm	-130.1 dBm	+3.5, -2.9
15.	Carrier APC Noise Bandwidth ($2B_{LO}$ = 12 Hz)	+10.8 dB	+10.8 dB	+0, -0.5
	Carrier Performance			
16.	Threshold SNR in 2 B_{LO}	+6.0 dB	+6.0 dB	--
17.	Threshold Carrier Power	-158.9 dBm	-158.8 dB	--
18.	Carrier Performance Margin	+20.3 dB	+28.8 dB	+3.5, -4.4
	Data Performance			
19.	ST/N/B Required	+7.0 dB	+7.0 dB	+0, -1.0
20.	ST/N/B Achieved	18.4 dB	+13.1 dB	--
21.	Data Performance Margin	11.4 dB	+6.1 dB	+3.5, -5.4

a. Maximum data rate or bandwidth

b. Maximum communication distance for a given data rate or bandwidth

c. Ground antenna gain

d. Ground receiver

e. Type of modulation and detection

f. Spacecraft antenna gain

For the Multiple Satellite Program, items b through e above are fixed by the orbital characteristics of the satellite array or are determined by the characteristics of the STADAN network. Thus, items a and f are the two major items which can be varied.

The choice of a suitable transmitter output power also is limited to certain quantum steps, particularly when one is constrained by the necessity of utilizing non-development hardware. For example, applicable transmitters are presently available with output powers of 1, 2, and 2.5 watts. Currently, a lower practical limit for S-band transmitter power appears to be about one watt. At an efficiency of 15 percent, a one-watt transmitter would consume about 6.6 watts in input power. Output powers less than 1 watt would suffer from an efficiency standpoint, since considerable power loss is experienced in the frequency multiplier stages, which are, for any power, constant. Conversely, a two-watt transmitter could have an efficiency as high as 20 percent, thus requiring an input power of 10 watts.

It is desirable to use the lowest power level possible to maintain communications at a given data rate or bandwidth, at the maximum distance required; however, very low power levels are not particularly advantageous, since devices are not available or are inefficient, at very low power levels. The first step is, then, to select a power level and calculate the link performance, using a spacecraft antenna with a gain of 0 dB. For this purpose, keeping in mind hardware limitations, a transmitter of one-watt output was selected. The results show that, for a slant range of 80,000 statute miles, and a nominal data rate of 1280 BPS, a performance margin of 5.1 dB is obtained. It is desired to maintain a minimum performance margin of 6.0 dB, thus the situation is marginal, but not entirely unacceptable.

The limiting case does occur when the data rate is raised. For tape recorder playback, it is desired to transmit data at a rate 24 times that of the real-time rate, or 30,720 BPS. A similar analysis for a 0 dB antenna shows that the performance margin at this higher data rate is reduced to 0 dB at a slant range of about 30,000 statute miles. Orbital considerations and STADAN viewing times dictate that, in order to transmit back the recorded data, it is mandatory to begin transmission at about this range. Therefore, it is certainly clear that this is indeed a limiting case, and that additional gain must be provided.

Since the critical parameters of the STADAN network are fixed, one may either (a) increase transmitter power to provide a gain of 6 dB, or (b) provide a spacecraft antenna with a gain of 6 dB. If alternative (a) is selected, a transmitter of four-watts output capability is required. Assuming an efficiency of 25 percent (at S-band), total input power of 16 watts would be required which is not supportable by the available spacecraft power. Therefore, an increase in transmitting power is ruled out, and attention must be directed to alternative (b) which was, in fact, selected.

With a transmitter power of one-watt and a spacecraft antenna gain of 6.4 dB, which appear in the present design, communications can be maintained over all portions of the orbit at the nominal real-time data rate of 1280 BPS, and adequate coverage is obtained to playback the recorded data at a rate of 30,720 BPS. In addition, another mode of operation could be provided if desired to playback the tape recorder at or near apogee at a lower rate. To maintain a 6.0 dB margin at apogee, a data rate of about 5000 BPS could be supported; addition of this mode could afford the opportunity of playing back recorded data at any orbital region.

4.9 ANTENNA SELECTION

Communication and tracking links under consideration for the multiple satellite system include the following:

a. Command link to each satellite

b. Data link from each satellite

c. An up and down link to each satellite for the GRARR system

d. Command link to the pallet

e. Tracking beacon on the pallet

Ground facilities supporting the multiple satellite mission are assumed to operate at S-band.

Performance requirements for the antennas derived from system operation and data transmission considerations are summarized in Table 13.

Data Transmission Antenna

Several types of antennas can provide a single beam normal to the spin axis and uniform in azimuth. Multiple rectangular horns mounted around the girth of each satellite are favorable from the structural and moment of inertia standpoint. However, this location precludes use of the antenna system while the satellites are in the pallet. The alternative is to place the antenna above the satellite on the spin axis. Two antenna configurations were investigated for this location. The first is a bi-conical horn and the second is a collinear array. The bi-conical horn design required to achieve the desired beamwidth of 25 degrees is excessively large in relation to the satellite, and requires a considerably larger support structure. In addition, the size is also unfavorable with regard to electrical performance. Optimum gain can't be achieved with dimensions of the order of one wavelength. The estimated gain of the bi-conical antenna for a 25 degree beamwidth is 5 dB, which is approximately 1 dB less than the required 6.4 dB. The collinear array, with three elements, fulfills both the electrical and structural requirements with minimum weight. Support for the array can be provided by a thin sleeve

Table 13

ANTENNA PERFORMANCE REQUIREMENTS

	Satellite Data Transmission	Satellite RARR Transmission	Satellite Command Reception	Satellite RARR Reception	Pallet Track TLM Transmission	Pallet Command Reception
Frequency Range (GHz)	2.2 - 2.3	2.2 - 2.3	1.7 - 2.3	1.75-1.85	2.2 - 2.3	2.2 - 2.3
Gain Relative to Isotropic (dB)	6	6	0	0	0	0
Beamwidth (deg)						
Elevation*	25	25	90	60	-	-
Azimuth**	OMNI	OMNI	OMNI	OMNI	-	-
Coverage	-	-	-		Lower Hemisphere	Lower Hemisphere
Power Capability (Watts)	5	5	-		0.1	-

NOTE: *Elevation angle is measured in relation to spin axis.

**Azimuth angle is measured in plane normal to spin axis.

of dielectric material, although the effect on performance at these frequencies would require further investigation. A self-supported array would have a larger diameter which might impose difficult element excitation problems. A sketch of the antenna for the satellite is shown in Figure 28.

Range and Range-Rate Downlink Antenna

The range and range-rate transmitting antenna requirements are similar to the data link antenna and, therefore, consideration was given to the use of the same antenna for both transmitters. The telemetry transmitter and the GRARR transponder cannot be operated simultaneously because of power limitations. Each transmitter can be switched into the antenna on command. The switch can be arranged to be normally closed to the data transmitter. With this technique, there is no restriction on operating frequencies; however, from a reliability standpoint, the switch is quite undesirable. An alternative approach is to diplex both transmitters into the collinear array. Because isolation requirements between transmitters are minimum, a frequency separation of 50 MHz is ample. A diplexer consisting of a three-port circulator and low-pass filter will fulfill this isolation requirement and is highly reliable.

Range and Range-Rate Uplink Antenna

This antenna is required to operate some 400 MHz below the transponder transmitting antenna and the gain requirement is considerably less. A separate antenna having the pattern of a dipole aligned with the spin axis is adequate for this purpose.

Satellite Command Antenna

Requirements similar to the GRARR system exists for the command receiving antenna. However, the operating frequency, at this point, is quite indefinite, since it could be anywhere in S-band, implying a range of 1.7 to 2.3 GHz. Both receiving requirements can be met by a common broad-band dipole antenna mounted on the spin axis above the collinear array.

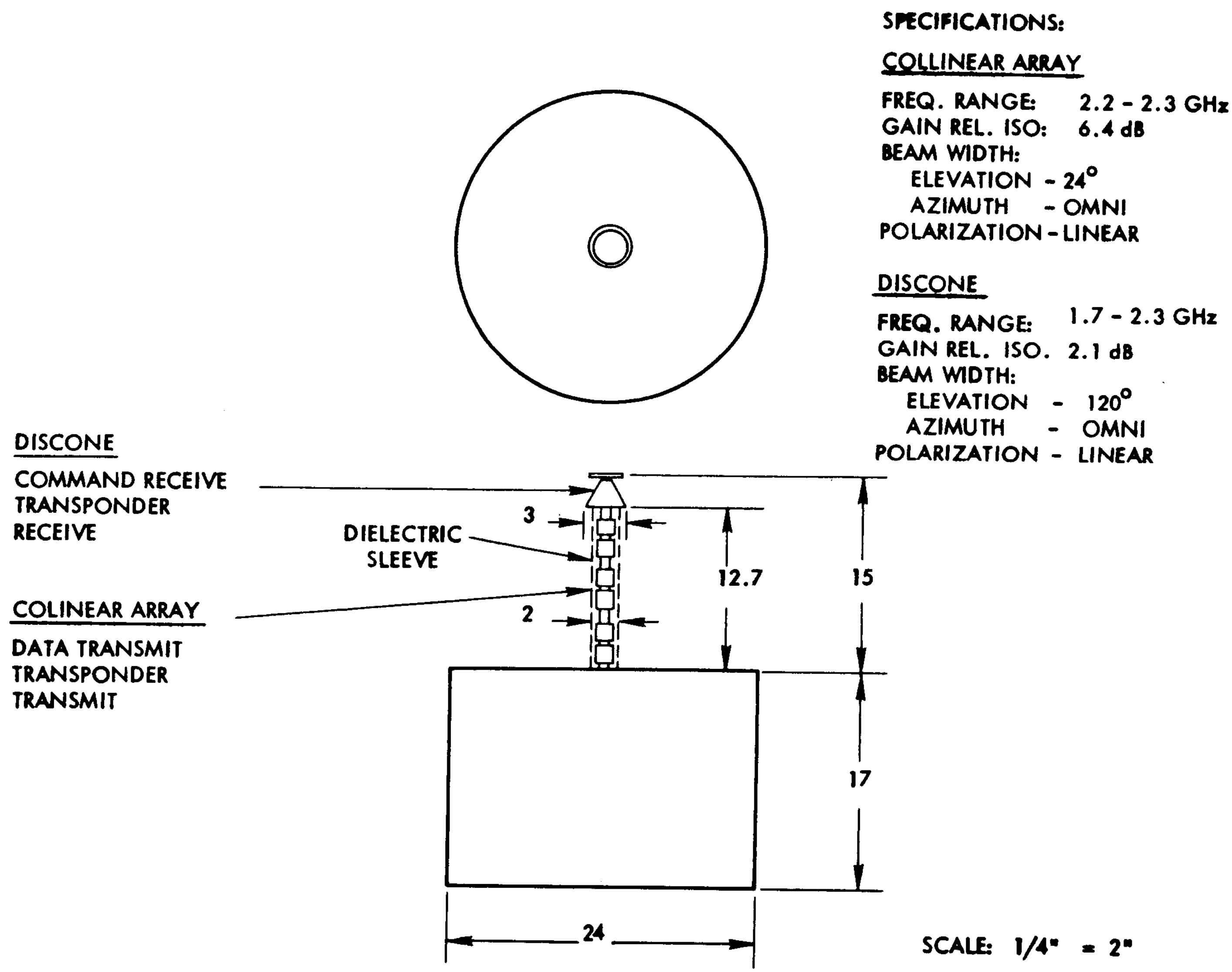


Figure 28. Satellite Antennas - Selected Design

Since the command receiver operates continuously, a diplexer is needed to provide isolation between receivers. The choice here is between a hybrid and a multicoupler operating band pass filters.

Receiving antenna design, even though it could be pursued more profitably when the command frequency assignment is made, was carried to the point that a broad band discone design (as shown in Figure 28) appears acceptable. The coaxial relationship and physical separation between antennas will yield about 20 dB of isolation. If the overall height of the antenna system becomes critical, modifications of the cone to increase the ground plane effect may be required within the constraints imposed by broadband design. This discone antenna is fed by a coaxial line within the collinear array. The feed lines are separated inside the satellite by means of a quarter-wave stub in the transmitting line to maintain a very high level of isolation..

Pallet Tracking Beacon Antenna

The most stringent requirement on the tracking beacon antenna(Fig 29) is broad coverage. A continuously radiated signal for pallet azimuth and elevation angle determination and for training the ground command antenna on the pallet is needed prior to the time the fan-beam antenna on the satellite can be used. The lower portion of the pallet will be oriented in the general direction of the Earth; therefore, an antenna with approximately uniform coverage over the lower hemisphere is required. The antenna configurations with this property are quite limited. At S-band, the antenna must be separated physically from the pallet to prevent shadowing and multiple lobes. A telescoping boom, however, is less reliable than the hinged boom, but it can be deployed along the spin axis. On the other hand, the hinged boom will be offset from the spin axis. With a slow rotational speed, the offset position is not detrimental to proper tracking. Indeed, the analysis given in a previous section with respect to doppler shifts indicates that such shifts do not pose any apparent problem. In addition, the physical arrangement of the hinged boom lends itself to convenient stowage and deployment. The antenna element at the end of the boom must be polarized transversely to the line-of-sight to the ground station over the lower hemisphere. This requirement is

SPECIFICATIONS:

FREQ. RANGE: 2.2 - 2.3 GHz

GAIN RELATIVE TO ISOTROPIC: 0 dB

COVERAGE: APPROXIMATE UNIFORM OVER HEMISPHERE

POLARIZATION: CIRCULAR ON ROLL AXIS
LINEAR IN ROLL PLANE

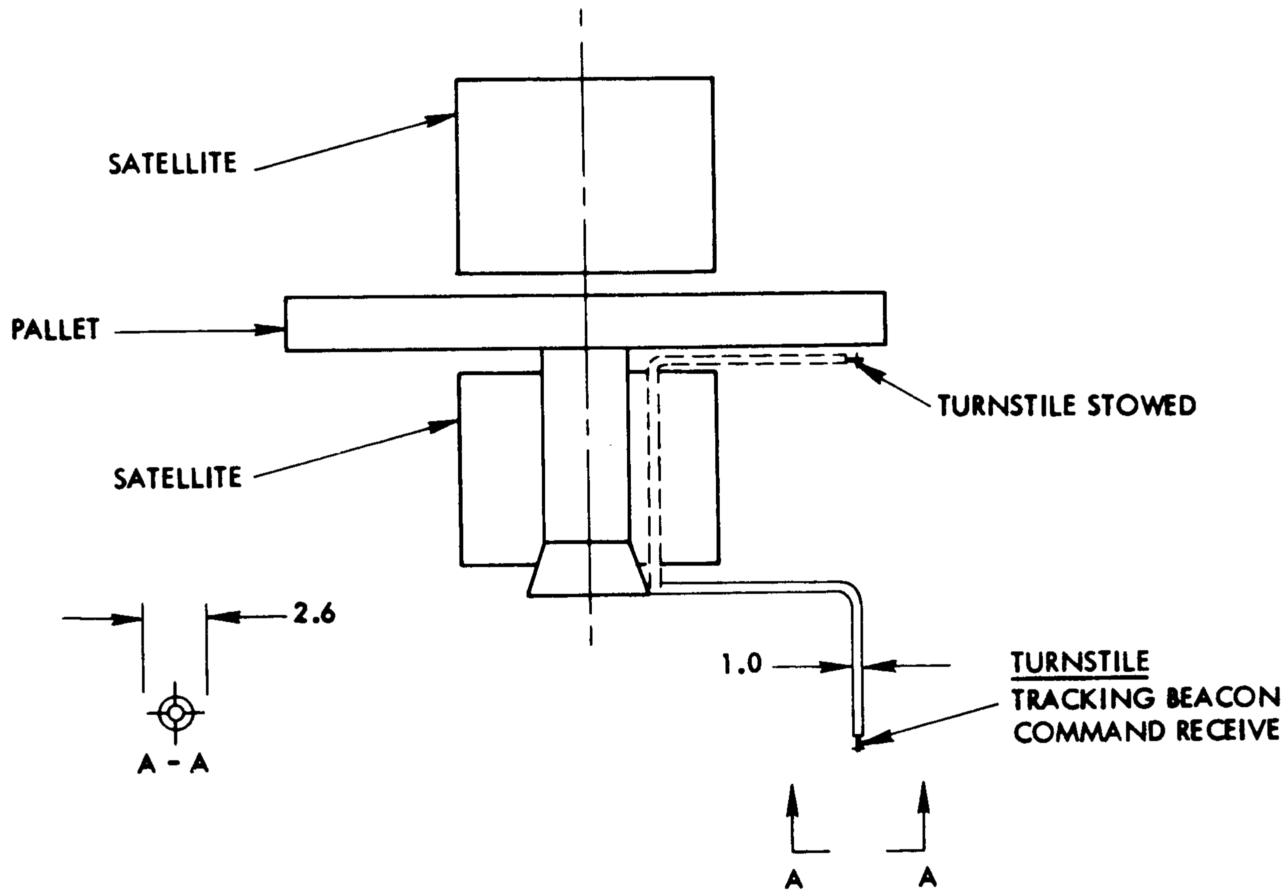


Figure 29. Pallet Antenna - Selected Design

fulfilled by a pair of cross-dipoles fed in quadrature. The polarization is circular along the spin axis and linear in the plane normal to the spin axis and the pattern is uniform in the same plane. If the ground antenna is circularly polarized in the complementary sense, the coverage will be uniform within 3 dB over the hemisphere.

Pallet Command Receive Antenna

Since the requirements for the command receive antennas are similar to those for the beacon antennas, consideration is given to using the same antenna for both functions.

Simultaneous transmission and reception over a single antenna requires a high degree of isolation between transmitter and receiver. If the operating frequencies are separated by several hundred MHz, most of the isolation may be obtained with filters. By maintaining the beacon frequency above the receive frequency, harmonics are eliminated as a source of interference. A high pass filter in the transmitter lines reduces residual noise and other spurious signals to a level compatible with the receiver threshold.

An efficient means of coupling the antenna to the transmitter and receiver is a four-port circulator. If the VSWR of the antenna is below 1.2:1 at the transmit and receive frequencies, the circulator will provide an additional 20 dB isolation with an insertion loss of 0.3 to 0.5 dB. Depending on final command and beacon frequency assignments and command receiver characteristics, the single antenna approach, because of its higher reliability and lower weight, is selected for the pallet command and tracking antenna.

4.10 MAGNETIC TAPE RECORDERS

The magnetic tape recorder forms a critical element of the multiple satellite communications subsystem due to:

a. The large amount of data acquired over the two-day orbital period.

b. The high reliability necessary for four satellites over the six-month system lifetime.

c. The requirement for extreme satellite magnetic cleanliness.

Thus, an investigation of tape recorder technology and availability was carried out for units applicable to the multiple satellite mission[20].

[20] See Appendix XVII.

Endless Loop Recorders

For this application, the principal objection to the use of endless loop recorders, aside from an undistinguished history, is the limitation on the tape capacity. The most recent of such machines cannot be expected to hold more than 300 feet of tape. Experiments which have been tried in the past using much longer tape loops have been markedly unsuccessful, because:

a. The support required for the tape loop as it rotates results in an enormous proliferation of rotating members

b. The frictional problems which are encountered because of the necessity of the tape to slide layer upon layer

c. The inability to maintain proper tape guidance

Hence, to summarize, because of (a) the previous poor history of operational reliability of endless loop tape recorders, plus (b) the limited data storage that can be achieved with the best of these recorders, which results in a high duty cycle for the STADAN stations; the use of endless loop tape recorders is not recommended for the multiple satellite mission.

Reel-to-Reel Recorders

Another basic type of recorder which appears much more suitable for use in the multiple satellite mission is a reel-to-reel recorder. Many of these recorders have been built and used in programs other than scientific satellites because of the generally higher storage capacity required in such programs. For this application, reel-to-reel recorders having a data capacity somewhat in excess of 10^8 bits are available as off-the-shelf items, which can record the scientific data for a total of 24 hours without resorting to recorder playback at any point during the data acquisition mode. Thus:

a. The STADAN duty cycle is minimized

b. Communications system complexity is minimized and subcarrier operation of two data links is not required

c. All the desired scientific data is obtained without interruption

In spite of the fact that many reel-to-reel recorders are available, much care must be taken in the construction of such recorders to achieve the reliability required to meet the long lifetime for the Multiple Satellite Program. The mechanical performance of a tape recorder probably is most dependent on the bearing assemblies. High-class (six or above) bearings, properly lubricated, must be used throughout a recorder to insure maximum possible lifetime.

Most tape recorders which have been flown in satellites or deep-space missions, and all which are currently being built, utilize hysteresis-synchronous motors rather than dc motors, in spite of the somewhat higher minimum power consumption of an ac motor. DC motors have several major disadvantages, including: (a) arcing, causing radio frequency interference, and (b) rapid wear of the motor brushes, resulting in lifetimes of considerably less than 500 hours. Development of brushless-dc motors has for many reasons been relatively unsuccessful. Invariably the techniques used to eliminate brushes result in a complex device utilizing electronic controls whose reliability is low compared to that of the ac hysteresis-synchronous motor.

The magnetic tape which would be used in the recorder for this application would be of a new high-frequency, Mylar-base type primarily developed for use with instrumentation ground recorders. Such tape, however, does have limitations if stretched by the application of high torque loads. Generally, this would not be a problem in a satellite tape recorder because of the limited torque characteristic of the drive motors. The more important problem is the thermal stability question. Mylar-base tapes tend to exhibit sticking or decomposition if stored at temperatures in excess of 200°F in intimate contact with typical head materials. Therefore, care must be taken during the entire testing, storage and handling procedure not to exceed a temperature appreciably above this. In addition, the thermal control of the spacecraft should limit the maximum temperature to approximately 125°F to permit a suitable margin of safety. Since thermal control design is well established to limit temperatures below this, consideration need not be given to other experimental tapes utilizing H-film or of an entirely metallic structure.

The most important element in any tape recorder is the recording and/or playback heads. It is here that the transformation of signal energy from an electrical current to a magnetic field, and vice versa, is made. The performance characteristics of magnetic heads have shown considerable improvement over the past several years. Such heads, which are generally available of conventional design, are built of laminated sections of magnetic material.

There are two basic types of reproduce heads, the so called $d\varphi/dt$ and flux sensitive heads. For the multiple satellite mission, it is most probable that a conventional head of the $d\varphi/dt$ type would be used as opposed to the flux sensitive head. The conventional heads are much more readily available, and constructed of materials which are eminently suitable to meet the lifetime and accuracies required. The flux sensitive heads generally are much more developmental and involve either new techniques of head construction which have not been proven in space or require the use of additional electronics.

Recent Developments in High Density Recording

Most tape recorders for spacecraft use record digital signals at bit packing densities of no more than 2000 bits per inch. Such densities are used because of several reasons:

a. Limitations on the tape surface

b. Limitations of the record-playback heads

c. Non-optimum playback circuit design

d. Conservatism in design

Recently a prototype recorder based upon a standard flight-qualified spacecraft recorder has been developed which operates at a bi-phase bit packing density of 10,000 bits/inch. It is apparent that, pending successful further development of this approach, such a high density recording technique is certainly desirable for the Multiple Satellite Program. Basically, savings can be made in weight and power and operational reliability is improved since a recorder utilizing this bit packing density could record approximately 1.1×10^8 bits in a single pass on a single track without need for any reversing operations during the data acquisition or playback mode.

4.11 COMMUNICATIONS SUMMARY

A summary of the multiple satellite communications capability, including data acquisition, downlink, command and tracking functions, is presented below. Figure 30 shows the periods during the orbit when data is recorded and transmitted.

Data Acquisition

a. The data acquisition rate is 1280 BPS for six instruments, aspect sensing and housekeeping.

b. The recorder storage capability of 1.1 x 10^8 bits allows storage for 24/hours/orbit, which covers the region from 8 to 18 R_e on both legs, as shown in Figure 30.

c. Provision for a real-time data acquisition and transmission mode is included.

Downlink

a. A one-watt S-band transmitter and a 6.1 dB collinear array antenna are used for the downlink.

b. Readout of a 24:1 playback ratio,, or 30, 720 BPS, is accomplished during the orbital period from 7.5 R_e to 4 R_e on both legs. (See Figure 30.)

c. The STADAN duty cycle requirements for the record mode are: 1 hr/station/day, typical
Less than 6 hr/station/day, all cases.

d. No pallet downlink is provided.

Command

a. Omni command receive antennas are specified

b. Capability is provided for approximately 75 command functions per satellite, and 11 command functions for the pallet.

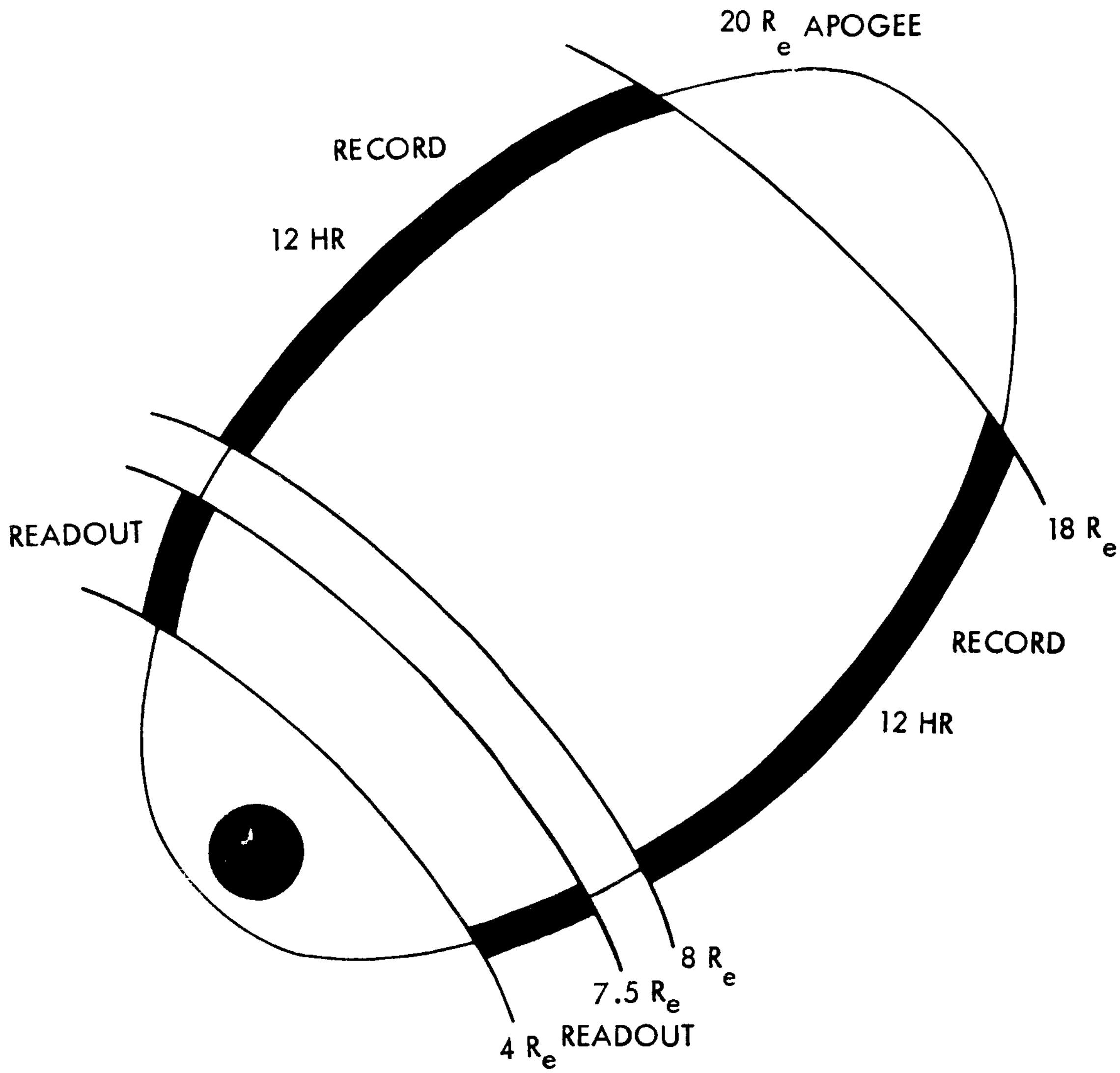


Figure 30. Data Record and Readout

Tracking

a. The GSFC range/range rate system will be utilized for tracking

b. An S-band beacon is specified for the pallet

c. The relative satellite positions can be established within $<$ 4 km, based upon an ephemeris maintained by 3 rangings per orbit from each of two ground stations.

Section 5

SYSTEM INTEGRATION

System integration analyses were performed throughout the duration of the program to guarantee compatibility of all subsystems. Satellite and pallet functional diagrams were prepared to assure subsystem functional compatibility. Integration criteria, system requirements, limitations, and design goals were established for all major subsystems. Included were the criteria and design limitations due to instrument integration, magnetics, reliability, electromagnetic interference, and environment. An evaluation approach was developed for subsequent use in comparing and selecting the most effective of alternately available components and subsystems. Support systems requirements were considered, emphasizing those aspects of support systems peculiar to the Multiple Satellite Program. A qualification and test philosophy for the Multiple Satellite Program was established. The system integration work performed during the program is summarized below.

5.1 FUNCTIONAL DIAGRAMS

Functional diagrams for the pallet and satellites were prepared to assure compatibility among all subsystems. Figure 31 represents the pallet block diagram. The major subsystems and their functional parts are identified. The figure shows the G-switch initiated functions, including satellite precession damper uncaging, antenna boom release and beacon. The attitude control pneumatic and electronic subsystems are included, and the command receiver and power system are also illustrated.

Figure 32 presents the satellite functional block diagram. The diagram shows the instrument's digital outputs multiplexed and fed to the tape recorder from which they are transmitted to the ground station. The engineering analog measurements are shown converted to digital form, multiplexed, recorded and transmitted. Command and tracking elements are shown including

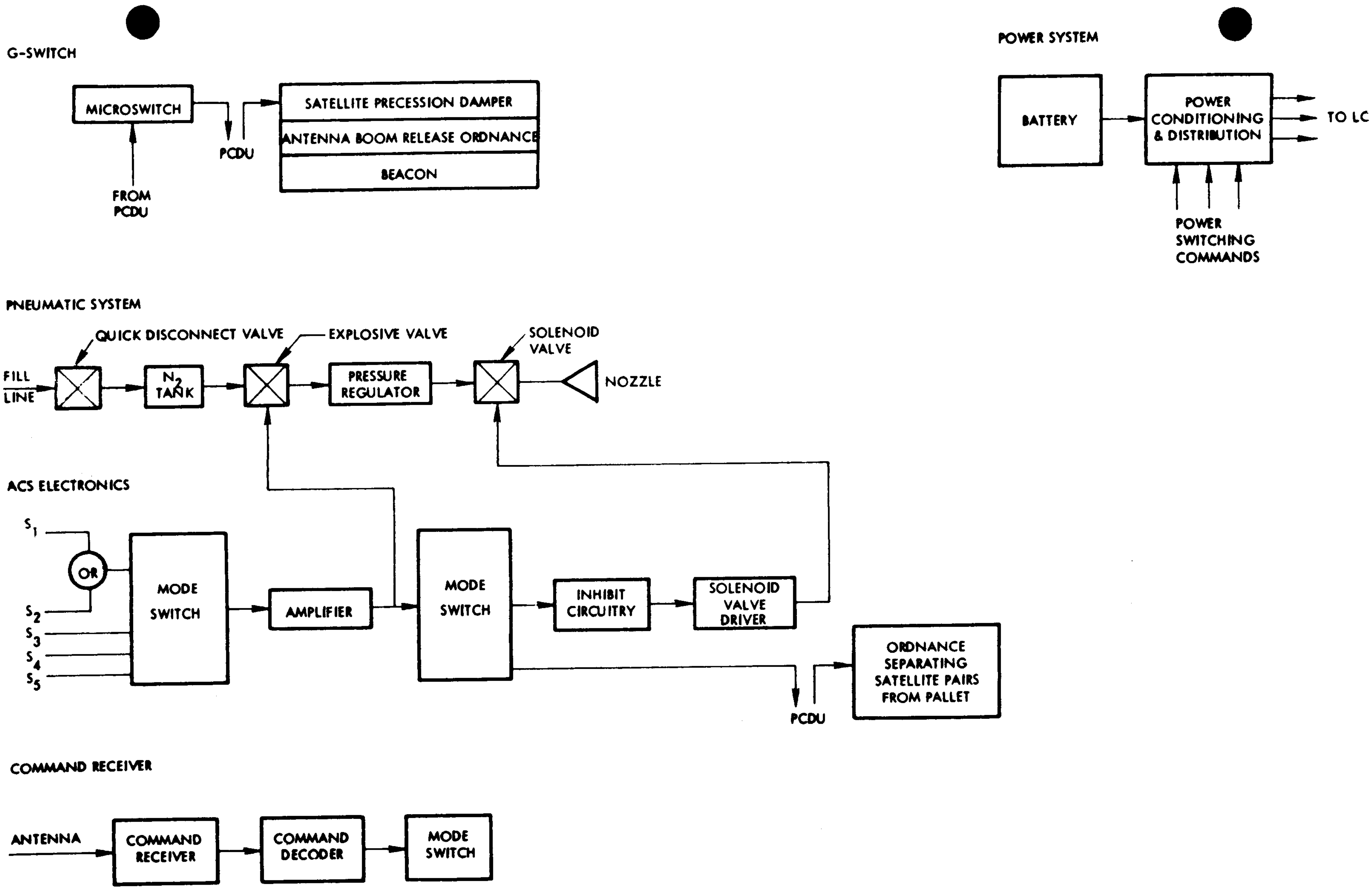


Figure 31. Pallet Block Diagram

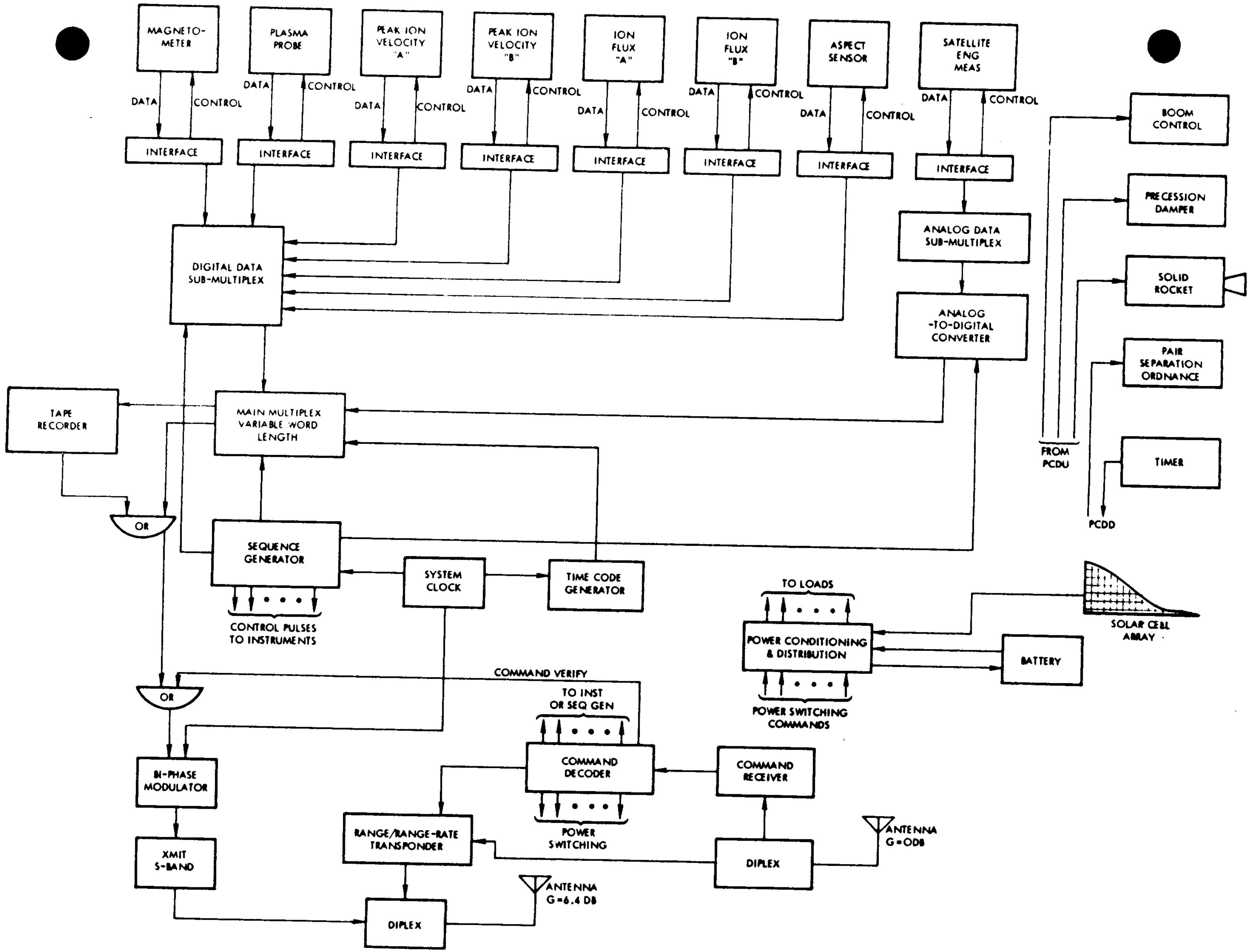


Figure 32. Satellite Block Diagram

the command receiver, decoder and transponder. The power system includes the solar cell array, power conditioning and distribution unit, and the battery. Each of the operations initiated by ordnance is shown, including magnetometer boom release, precession damper uncaging, pair separation from the pallet, and the out-of-plane separation solid rockets.

Consideration of these functional diagrams will serve as an introduction to the interaction of the various subsystems, and the system integration and design tasks.

5.2 INTEGRATION CRITERIA

Integration criteria, limitations and design goals due to the requirements for a magnetically clean and reliable spacecraft were evolved. In addition, criteria for electromagnetic interference control and for satisfactorily surviving the expected environment were established. These are given in 5.2.1 through 5.2.4 below.

5.2.1 MAGNETICS

One of the primary objectives of the multiple satellite is the systematic and continuous measurement of the magnetic field in interplanetary space, in the transition region, and within the magnetosphere. This magnetic field can be as small as 5 gamma in the interplanetary region, and the spacecraft must be able to detect this magnetic field with reasonable accuracy. If the magnetic field contributed by the spacecraft exceeds the field to be measured, but is predictable and steady, then measurement of the small interplanetary field would be possible. Detection of the interplanetary field, when the spacecraft field exceeds it, is generally not possible, however, since the spacecraft field is not predictable and steady within the accuracy required for the measurements. Therefore, it is necessary to maintain the spacecraft field (at the magnetometer sensor) at a level sufficiently lower than the field to be measured. This level has been specified as 0.5 gamma, which includes all causes (permanent and stray fields).

The magnetic field from a static dipole source falls off as $\frac{1}{r^3}$. The desired field at the magnetometer sensor could be achieved, no matter what the spacecraft field is, by locating the sensor far enough from the spacecraft to reduce the field at the sensor. However, practical limitations due to achievable boom lengths, etc, soon limit the separation distances between the sensor and spacecraft. Based upon preliminary estimates of the achievable magnetic cleanliness of the spacecraft, boom lengths of the order of six to eight feet will be required. Assuming 0.5 gamma at the magnetometer sensor and a six- to eight-foot boom, the spacecraft field limitation is 4 to 10 gamma (at three feet) distributed among the basic subsystems.

Magnetic design considerations should start with the component, subsystem and system layouts and materials selection to prevent design decisions which neglect magnetic considerations and, thereby, create future magnetics problems. A detailed description of magnetic design guidelines has been prepared, including a listing and description of non-magnetic materials, a guide to preferred wiring practice, and shielding considerations.[21] Shielding is generally not a recommended practice; elimination of the magnetic field or its compensation is preferred.

Subsystem and component location and orientation within the spacecraft (from a magnetic standpoint) should be considered, beginning with the preliminary layouts. In general, "dirty" parts, or those with large currents (tenth of an amp to amps) which will produce large stray fields, should be located as far from the sensor as possible, on the side of the spacecraft opposite the sensor boom. Those components drawing milliamps may be located in the spacecraft region close to the sensor boom. In addition, spinning vehicles may suggest special location preferences. As the program progresses into design layout and the magnetic performance of the various components becomes available, more definitive location specifications can be provided.

Preliminary subsystem magnetic specifications and qualification procedures have been prepared for the multiple satellite.[22] Essentially, the

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[21] See Appendix XVIII

[22] See Appendix XVIII

design goal for each subsystem which will produce a magnetic field was set at 2 gamma (at three feet) after 25-gauss exposure and 0.5 gamma in the depermed condition. Stray field allowances, dc through 25 cps, were set at 0.2 gamma at three feet. It is not possible to specify in detail the magnetic requirements for each subsystem at this time, but the above design goals are based upon the over-all spacecraft requirement and do represent potentially achievable subsystem characteristics. As more data on subsystem capability becomes available these preliminary goals will be revised.

5.2.2 RELIABILITY

The mission of the Multiple Satellite Program dictates that the initial design be characterized by a high level of inherent reliability. This objective has been and will continue to be a prime consideration in performing reliability trade-off studies and making hardware and configuration selections. The reliability requirement will be met through the utilization of proven state-of-the-art components with known flight qualification status and failure-free test histories, wherever possible. This basic feature will be complemented by maintaining design simplicity, prudent usage of derating and safety factors, selective application of parallel redundancy in critical components, and the imposition of mandatory reliability design practices.

A basic system reliability design goal has been specified. This goal requires that the system of four satellites function over a period of three months at a reliability of 0.70 or better. A preliminary reliability budget has been developed from the implications of this design goal, together with the interpretation of mission success relative to specific hardware operation. This budget is presented in summary form[23] in Table 14. The starred items represent subsystems which may present potential difficulties in meeting the individual goals. The reported design goal for the THORAD by its manufacturer is 0.85 which is less than given in Table 14. The Thor Delta is currently operating at a reliability of

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[23]See Appendix XIX for details

Table 14

ABREVIATED RELIABILITY BUDGET

	Quantity/ System	Reliability Goal	
LAUNCH			.9673
Boost Vehicle (Thorad)*	1	.9705	
Multiple Satellite System during boost	1	.9967	
PALLET			.9018
Structure & Spin-Off Release	1	.9883	
Thermal Control	1	.9975	
Command Rel. and Antenna	1	.9761	
Battery & Power Conditioning	1	.9648	
Attitude Control	1	.9713	
SATELLITES			.8027
Structure	4	.9971	
Boom*	4	.9850	
Thermal Control	4	.9967	
Solid Motors	4	.9950	
Data System*	4	.9348	
Power System	4	.9478	
Precession Damper	4	.9950	
Instruments	4 sets	.9348	
	Total		.700

*Indicates Stringent Reliability Goal

about 0.93 which is also less than the budgeted amount. Other subsystems which may present difficulties in meeting the design goal include the magnetometer boom, and the data handling and storage subsystem.

5.2.3 ELECTROMAGNETIC INTERFERENCE CONTROL

Beginning early in the design phase and continuing throughout the Multiple Satellite Program, attention must be given to electromagnetic interference (EMI) produced by the spacecraft will be controlled to eliminate undesired malfunctioning of all electronic and electrical subsystems in, or associated with, the spacecraft. This requirement applies to the entire frequency range of the installed subsystems, including operation with their installed antennas when performing their intended radiation or reception pattern.

Basic EMI control features, such as grounding, bonding, shielding, packaging and cable design, will be considered during future design phases. Considerable previous work has been performed and documented on the subject of EMI control.[24] A complete design guide on specification of EMI control criteria is not possible, or warranted within the scope of the current study; however, the designers must be aware of its importance and the existence of the referenced criteria documents. Particular attention should be given to the unique requirements of this program, i.e., multiple satellites operating in close proximity, when selecting frequencies and bandwidths, and establishing EMI control requirements.

5.2.4 ENVIRONMENTAL CRITERIA

A summary of applicable environmental criteria was established to suggest minimum basic test levels for environmental testing of prototype and flight models of the multiple satellite.[25].

Included in the launch vehicle environment considerations are the expected vibration environment and the resulting design qualification and flight acceptance specification. The angular acceleration due to spin-up has been specified with the associated balance requirements. Detailed balance requirements for the spacecraft interface with the booster and for the different satellite/pallet combinations have been defined, and design load factors based upon the boost vehicle acceleration have been specified on a preliminary basis.[26]

The environmental conditions to be encountered by the satellites in the specified orbit have been reviewed, including the expected radiation levels due to trapped and solar radiation, meteoroid flux rate densities and the expected

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[24] See Appendix XX for references

[25] See Appendix XXI

[26] See Appendix XXI

thermal radiation. No unusual environmental conditions are expected to be encountered for the multiple satellite mission.

5.3 EVALUATION CONSIDERATIONS

A procedure has been established for evaluating candidate multiple satellite subsystems and components on a cost-effectiveness basis.[27] Relationships have been developed for cost-effectiveness in terms of data value; satellite, pallet and launch vehicle reliability; and cost; the optimum allocation of excess weight is also discussed. Influence coefficients to be used in evaluating alternative subsystems have been evolved. These influence coefficients can be used to establish the relative effects on the total program of the subsystems' cost, reliability, and weight. The use of these influence coefficients allows consideration of candidate methods and/or subsystems by establishing their specific effect on the total program. An example comparison of three prospective aspect sensors is presented to illustrate the method. The approach can be used in future design phases, when better subsystem cost, weight and reliability data becomes available, to aid in selecting the makeup of the best over-all multiple satellite system.

5.4 DEVELOPMENT SUPPORT CONSIDERATIONS

A preliminary support systems study was conducted to (1) determine broad system interface areas and to provide data for input to future program schedule and cost analyses; and (2) to identify systems problems peculiar to the multiple satellite mission, requiring more detailed investigation. The descriptions are preliminary and may change as the design of the spacecraft progresses.

To establish the equipment and operations needed in support of the Multiple Satellite Program, it is necessary to consider the entire program from the start of development through initial orbital operation. Thus the support systems required to assure the success of the multiple satellite missions must be divided among the following categories:

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[27] See Appendix XXII

- Fabrication and Assembly Equipment
- Ground Support Equipment
- Facilities Requirements

Fabrication and assembly equipment unique to the multiple satellite include pallet/satellite alignment equipment, a spin-separation test fixture, transporter and handling equipment and fixtures, magnetic cleanliness checkout facilities and equipment, and console elements peculiar to this mission.

Ground support equipment unique to the multiple satellite mission is expected to consist of a satellite evaluation van used during the prelaunch and launch support tests at the launch site. The van will contain a complete checkout console in addition to telemetry receiving equipment and recorders. This facility should provide a capability for checking both the satellites and the pallet, completely independent from the range facilities.

Based upon preliminary investigations, it appears that existing facilities will generally be adequate for the Multiple Satellite Program, with the exception of the special spin-separation test equipment and instrumentation.

5.5 TEST PHILOSOPHY

A preliminary test philosophy for the Multiple Satellite Program has been developed. The test program is aimed at the development and design verification of the multiple satellite through component, subsystem and system tests. These tests will encompass component acceptance tests, component and system qualification tests, and flight acceptance tests. Wherever possible, components will be procured as qualified items requiring only functional acceptance testing; however, some components must be developed for the multiple satellite application within the state-of-the-art. Such components as the magnetometer boom; ACS tankage, nozzle, and plumbing; and satellite structures and spin-off mechanisms fall into this category.

The only unique test requirements identified for the Multiple Satellite Program relate to (1) the development of the pallet spin-separation subsystem, and (2) the requirement for qualification to the stringent magnetics specification.

5.6 OPERATIONAL SUPPORT CONSIDERATIONS

The operational support system includes the facilities and equipment required to provide an integrated support network for the Multiple Satellite Program from launch operations and space flight operations to data acquisition, display and reduction. The STADAN ground stations and the NASA Communications System (NASCOM) have been designated to support the operational phase of this program; Goddard Space Flight Center is the focal point of these networks. Thus GSFC will logically be responsible for final coordination of the launch station, the overseas tracking and data acquisition stations, and the various computation and control centers.

The following operational plan must, therefore, be considered only as conceptual and suggestive in nature:

Launch Phase - Launch of the multiple satellite system will be accomplished at the Eastern Test Range (ETR). Checkout and tracking will be carried out using existing range equipment, supplemented where necessary by Multiple Satellite Program GSE. Use of the JPL Space Flight Operations Facility (SFOF) as the control center is suggested, with links to the test range via existing data/voice lines.

Deployment and Initial Orbits - During the deployment sequence and the satellites' initial orbits, real time contact with the satellites will be monitored by the control center at JPL via overseas stations. Commands will be generated at the JPL control center and relayed to the satellites via outlying stations utilizing existing NASCOM high data rate lines. Tracking and data information will be relayed from the outlying stations to the control center over the existing cables. Although JPL is suggested here as the initial control center, the actual selection will be based on capabilities and work loads, and must be confirmed and agreed upon by personnel at Ames Research Center, Goddard Space Flight Center and JPL.

Operational Phase - After the satellites enter their normal operational sequence, pre-planned data acquisition programs will be initiated at each STADAN station and real time commands relayed to the satellites from these stations. The control center for the operational phase would probably be switched to GSFC, with tracking and data information placed upon magnetic tapes at the stations or the control center, and delivered to ARC for reduction purposes. If limited real time monitoring is required at ARC, a high data rate line could easily be provided via the existing JPL links.

Section 6

SYSTEM DESIGN

Based upon the functional requirements of the preceding sections, the design effort provides a feasibility verification design for the pallet and satellites. The design results include configuration and weight studies; analyses and recommendations for thermal control design; design of satellite separation mechanisms; consideration of alternate magnetometer boom designs; determination of power requirements and specification of the power system design; and a component availability survey.

6.1 CONFIGURATION STUDIES

Structural envelope and component arrangements have been developed which will meet the environmental constraints and satisfy the operational requirements of the scientific instruments and the ancillary support equipment, within the payload weight and volume constraints of the launch vehicle. The simultaneous consideration of the functional and operational interfaces between subsystems results in geometrical arrangements of components and the supporting structures and mechanisms which define the candidate configurations. Some aspects of the subsystem interactions can be analytically formulated while others are largely practical with solutions based upon experience. Preliminary study usually evolves several configurations which satisfy the overall system requirements in a general sense. A detailed comparative design study is then pursued to develop the best configuration. This procedure is complicated, for the multiple satellite system, by the existence of three sequential configurations: (1) the payload (pallet/satellite pairs assembly); (2) the satellite pairs; and (3) the individual satellites (with boom extended and retracted). The evolutionary design process leading to the selected design is delineated in Figure 33. In this figure, the various concepts considered and their growth to the final preliminary design are shown.

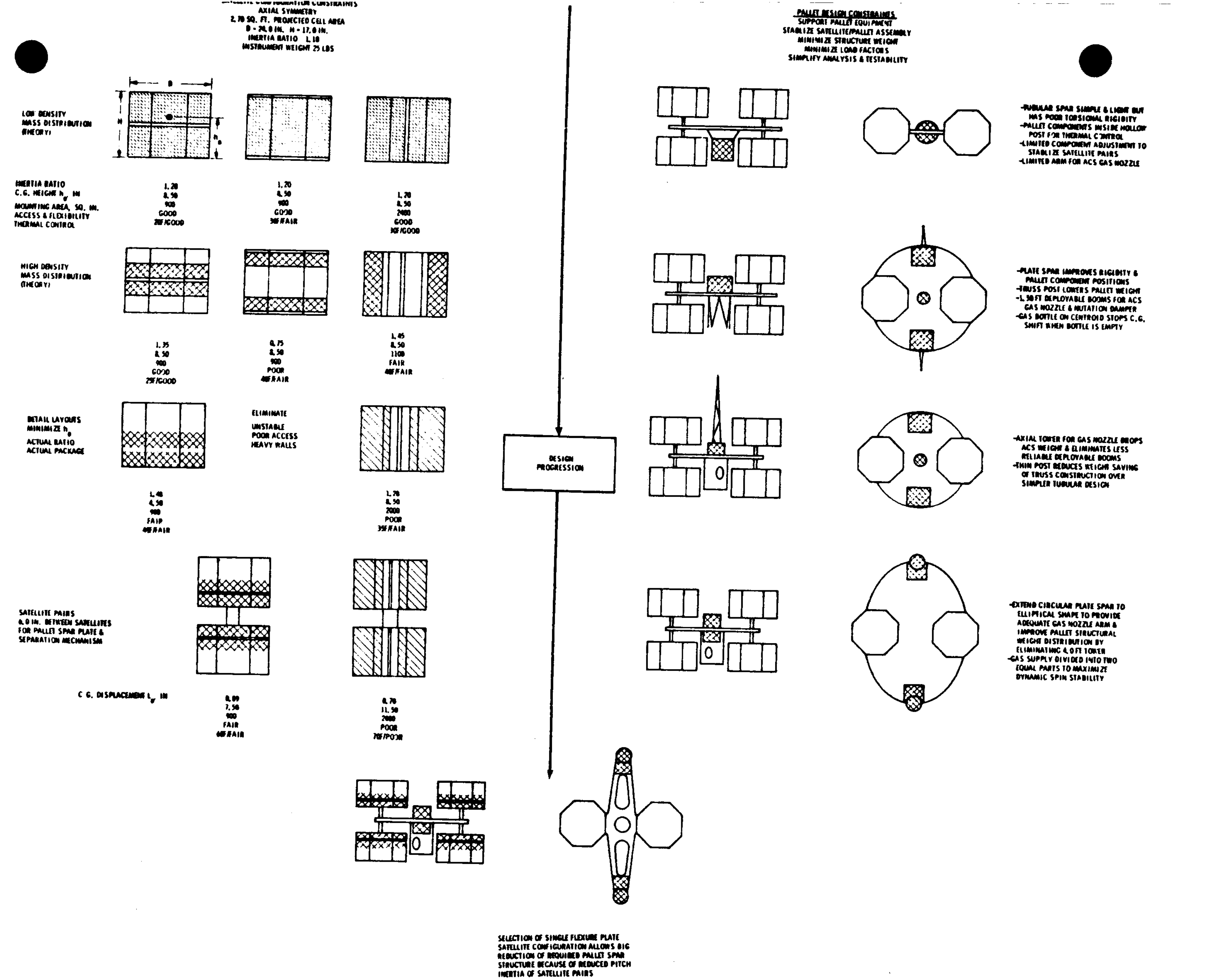

Figure 33. Design Evolution

6.1.1 SATELLITE CONFIGURATION

The selection of the external satellite configuration is established from the average power requirements which determines the projected solar cell area. The power requirements, therefore, determine the height and diameter of each satellite. Uniformity of power generation suggest an axially symmetric outline which in this case is octagonal, since flat sides provide simplicity of fabrication and the eight-sided arrangement permits perpendicularity of mounting surfaces.

The internal arrangement or configuration is related to the temperature control system; the mass distribution characteristics required for attitude stabilization;[28] component distribution to minimize the magnetic field at the magnetometer sensor; instrument packaging for checkout, viewing angles, etc.; and the requirement for as efficient a structural support system as possible. Thus the design of the internal configuration is more complex than the external.

As illustrated in Figure 33, three general approaches for component support were considered. These consisted of (1) a single transverse support plate with components fastened to both sides and the side walls fixed to the edges of the plate, (2) two transverse plates with components mounted to the inside surface of each plate and the solar panel side walls providing the structural separation between the plates, and (3) a cruciform of four vertical panels radiating from a central tube with components capable of being mounted on both sides of the four panels and the side walls attaching to the edges of the panels. In evaluating the development of each of the three alternatives, the four major criteria (thermal, balance, component mounting flexibility, and structures) were given more specific constraints. The thermal control was to be a completely passive system; attitude control requirements dictated a spin stability margin ($^{Ispin}/Ipitch$ - 1) of at least 10%; maximum possible area for mounting permitting access to the satellite center of mass (COM) was required for testing and balancing; and a structural design was required which would minimize bending.

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[28] See Appendix XXIII

6.1.1.1 SINGLE TRANSVERSE PLATE

The single transverse plate configuration presents some thermal problems, since the components are not mounted directly to a radiating surface. Here the conduction paths lead directly to the solar array where the surface properties are not variable. The thermal link to the enclosures is a radiative one which is relatively less efficient. This concept probably provides the best accessibility and mounting flexibility since direct access is available to all components without disturbing the solar array. The structural subsystem is of mixed efficiency. The solar side walls can be of light construction (about 0.50 lbs/ft^2) since they do not constitute a primary load path, however, the centrally supported transverse plate is subjected to considerably higher flexural moments than a similar plate supported at the periphery. One of the primary advantages of this geometry is that as its packaging density is increased, the location of the center of mass of the satellite can be varied by changing the vertical station of the plate. This has very significant spin stability influences on the satellite-pair geometry and on the overall payload configuration and pallet structure. Also, it was the primary factor leading to the selection of this configuration over the other alternates. A detail layout of this configuration is shown in Figure 34.

6.1.1.2 DOUBLE TRANSVERSE PLATE

The thermal conditions for the double transverse plate design are probably the best of all the configurations considered. The components are directly connected to the radiating surfaces which in turn can be thermally isolated from the solar array by means of non-conducting structural joint materials. This will allow cooler and therefore, more electrically efficient power generation. It will also permit more uniform internal temperature profiles. This situation improves as the packaging density is increased. Component accessibility is poor at both ends of the density spectrum since access is only through the solar panels. As the higher density arrangement is approached, the stability margin will go negative which violates the balance requirements. The structural subsystem will again have mixed efficiency since the side walls now constitute a primary load path and will

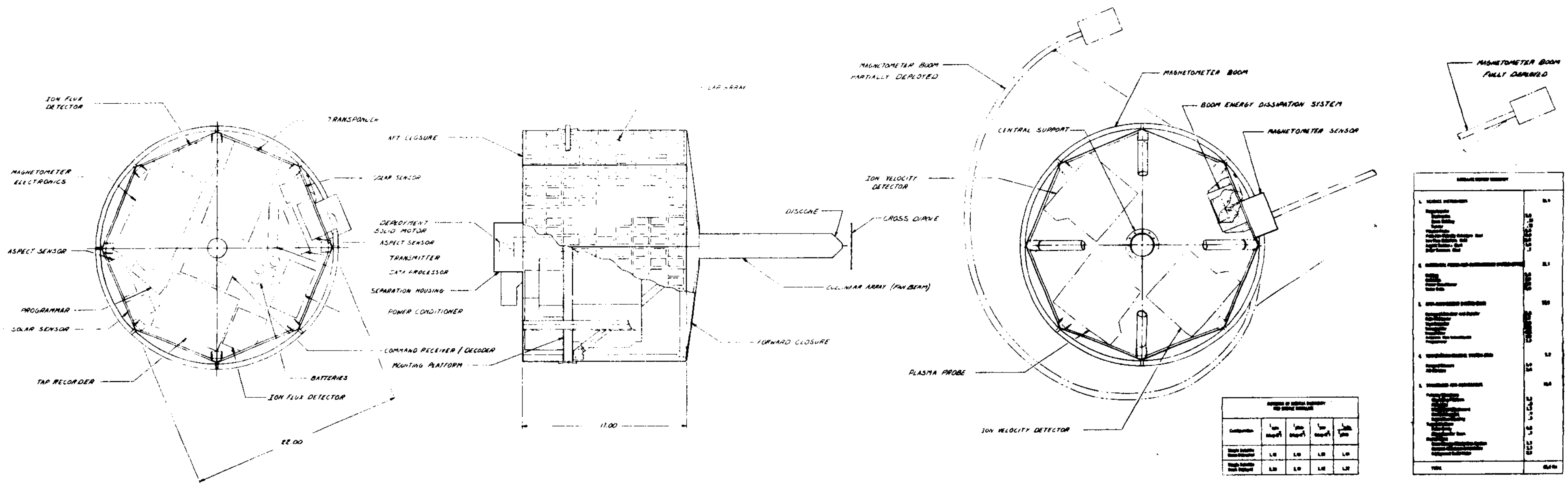


Figure 34. Satellite Configuration A

have to be heavier than in the single plate, while the plates themselves will be simply supported on the complete circumference. Since it is known that the packaging density will eventually be relatively high, this approach is considered inappropriate for detail development due to its balance inadequacies.

6.1.1.3 FOUR-PANEL CRUCIFORM

Thermodynamically, the four-panel cruciform geometry is difficult to assess accurately. This is because the conduction path can be either to the end enclosures or to the solar array panels for the uniform distribution. This design does provide the best configuration from the point of view of individual satellite stability, however, Mounting flexibility is optimum in that there is more than twice the available mounting surfaces for components. It also provides access to the center of mass (COM). However, once assembled, the only way to get at the components is through the solar array. This design will also provide the most efficient structural subsystem since the severest primary loads are all taken in shear and not in bending, the transverse loads being less than 1/2 the thrust loads. However, in the detail development of this approach it was learned that most of the components could not be subdivided into small enough units to take advantages of the semi-monocoque shell geometry. A detail layout of this satellite configuration is shown in Figure 35.

6.1.2 SATELLITE-PAIR CONFIGURATION

Before the best satellite configuration could be selected, the satellite design alternatives had to be considered in the paired configuration. (See Figure 33). The same four subsystem considerations pertain to the paired configuration as did to the individual satellite configurations. Spin-stability is still desired since this geometry must exist as a freely spinning body. A problem arises here since it is not possible to construct a satellite-pair for any of the individual satellite configurations that will have a positive stability margin. The requirement of accessibility is redefined in this case as being able to effectively handle, assemble, and adjust the mechanical interface that joins the two satellites together. The structural considerations

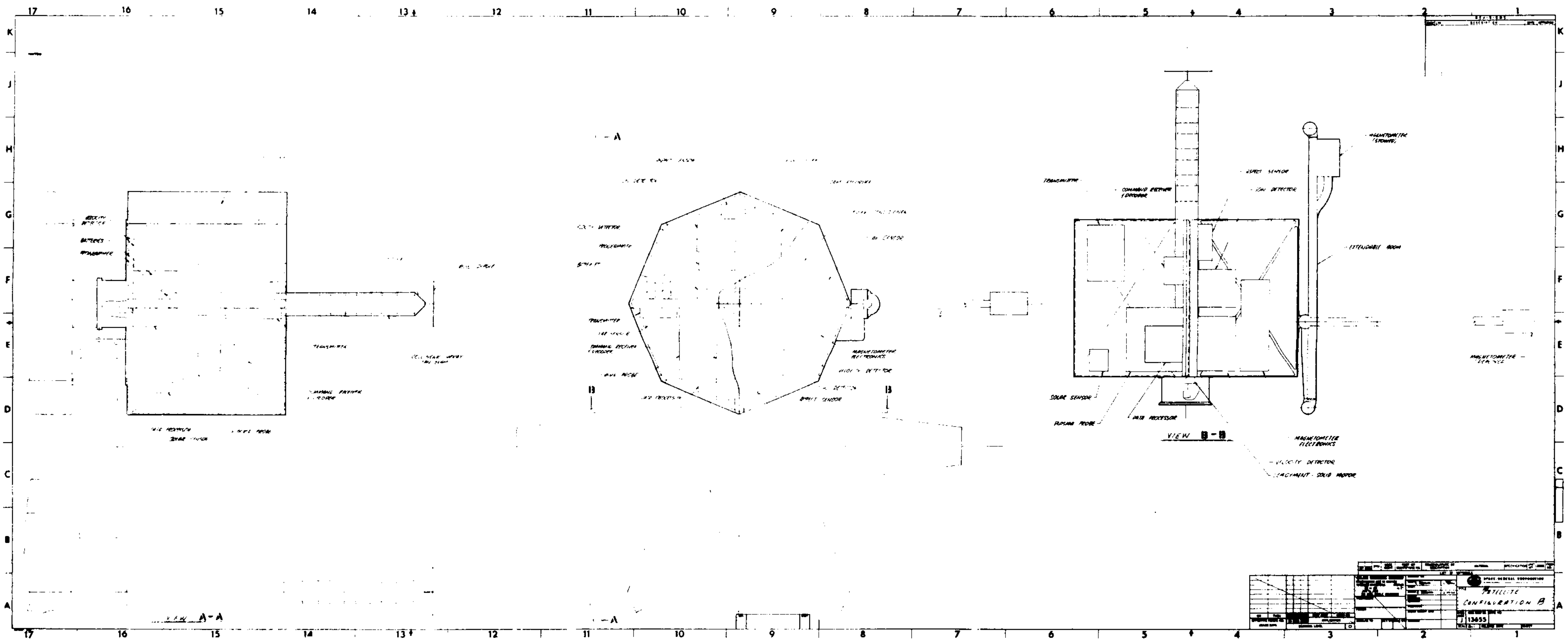


Figure 35. Satellite Configuration B

are likewise centered around the nature of the separation system that is designed. Thus the central design consideration focuses upon the separation requirements.

Design study indicates that for the payload weight to be maintained at less than the allowable 405 lbs without employing booms for stability,only the single plate satellite design with the plate displaced as close to the separation plane as possible will suffice. Sensitivity to pitching impulses is increased; however, these will be minimized by ensuring that the COM of the pallet lies in the same plane as the satellite pairs so that if one pair is spun-off slightly before the other, the only effect is to translate the system spin axis and not to introduce coning. Further, the design of the spin-off separation mechanism must take careful consideration of any effects (non-uniform axial thrusts, rough edges that will contact during separation, etc.) that can introduce pitching. The effect of increasing coning once the pairs are free is reduced by keeping any precesion dampers caged and by axially separating the pairs into the individual satellites as soon as possible after spin-off.

The satellite pair geometry therefore suggests selection of the transverse plate satellite configuration, with the plate shifted for balance and loads considerations. This imposes dynamic requirements upon the spin off separation mechanisms and upon the mass distribution for the pallet. It also requires that the satellites be close together when placed in pairs; thus the thinnest possible pallet spar cross section is advisable.

6.1.3 PALLET DESIGN

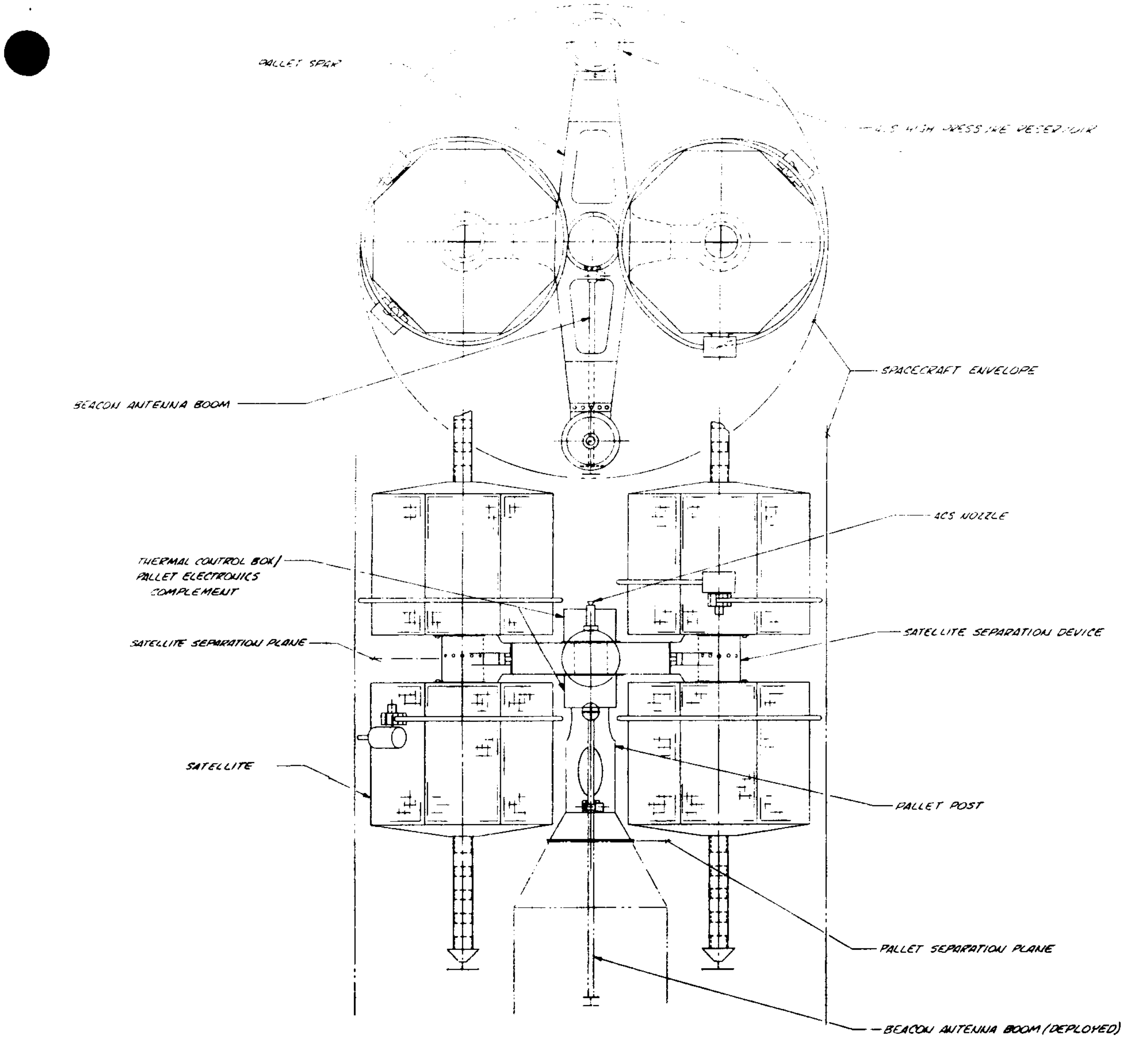
The pallet has several functional requirements as described by the operational sequence, Section 2.5. The pallet subsystems include an ACS consisting of control electronics, nozzles, cold gas reservoirs; a telecommunication command link; a tracking beacon; sun sensors; and associated data processing and logic electronics. Further, there are two mass distribution requirements: (1) the center of mass (COM) must be in the same plane as the satellite pairs; and (2) the masses must be located to establish a positive stability margin for the payload of at least 10%. A structural constraint also arose in that it is desirable to have the pallet spar cross section as thin as possible to minimize the instability of the pairs. The pallet design was evolved based upon these constraints.

The thermal control for the power dissipation components is provided by covering the components with thermal "boxes". Since the pallet components are more or less remote from the pairs, the conduction effects are minimized and the thermal input can be controlled by radiative surface coatings. No problems are anticipated in passively controlling the thermal limits for the pallet components. The requirement for spin stability has already been defined by the previously mentioned mass distribution requirements. Accessibility to the components and flexibility of mounting them present no serious problems since ample space is available on both sides of the pallet. The structural constraints are severe due to (1) the very large point loads and mass inertias at the points where the pairs attach and (2) to the fairly narrow diameter permitted for the pallet post.

As illustrated by Figure 33, the configuration progressed from a circular honeycomb plate spar into a cruciform or dumb-bell spar with edge-stiffened cut-outs. The total mass of the pallet (structure and components) was concentrated as much as possible on the tips of the cross-bar perpendicular to the axis joining the separation points of satellite pairs. Alternate post designs were considered, including a truss ultimately evolving to a honeycomb tube with elliptical cut-outs to reduce the weight.

The reduction of the attitude control system weight was considered by providing a long moment arm (tower) for the nozzle. The ACS nozzles were finally located at the tip of the spar; the total cold gas supply was divided into two tanks and also located at the spar tips to aid in system balance.

The pallet design and satellite mounting arrangement is shown in the payload drawing of Figure 36. The primary design constraint for the union of the pairs and the pallet (Figure 36) is centered about the spin-off separation mechanism. Some of the constraints upon this mechanism have already been discussed. The separation mechanisms, both for spin-off and the axial separation, are unique to the multiple satellite mission and are of critical importance to its overall success.



SUBSYSTEMS WEIGHT SUMMARY

	COMPONENTS	SUB-SYSTEM
A. EXPERIMENTS		342.4
Satellites (4 required at 85.6 lbs ea)	342.4	
B. DATA MANAGEMENT (DMS)		5.0
Command Receiver and Decoder	3.0	
Antenna Beacon	2.0	
C. ELECTRICAL POWER AND DISTRIBUTION (EPDS)		3.7
Batteries	2.2	
Power Conditioner	1.5	
D. ATTITUDE CONTROL (ACS)		24.8
Nitrogen Gas	11.0	
High Pressure Reservoir (2 required)	10.0	
Electronics	1.5	
Valves, Nozzles, and Plumbing	1.5	
Solar Sensors (3 required)	.8	
E. THERMAL CONTROL (TCS)		1.0
Surface Control Boxes (4 required)	1.0	
F. STRUCTURE AND MECHANISMS		22.5
Pallet Spar	13.0	
Pallet Post	3.0	
Satellite Separation Device (2 required)	5.0	
Beacon Antenna Boom	1.5	
TOTAL		399.4 lbs

MOMENTS OF INERTIA SUMMARY
FOR SATELLITE PAIRS AND COMPLETE SYSTEM

Configuration	I_{spin} (slug-ft^2)	I_{pitch} (slug-ft^2)	I_{yaw} (slug-ft^2)	$\frac{I_{spin}}{I_{pitch}}$
Satellite Pairs (Booms Retracted)	2.96	3.31	3.29	.89
Satellite Pairs (Booms Deployed)	5.82	4.74	5.72	1.22
Pallet System (Booms Retracted Without N_2)	31.70	26.85	12.17	1.17

Figure 36. Pallet/Satellite Configuration

150
~~427~~

6.2 SEPARATION MECHANISMS

The separation mechanisms may be divided into two specific areas. The first, and most difficult, is the spin-off separation of the satellite pairs. The second, and more conventional is the axial separation of the individual satellites. Each of these separation techniques has unique requirements. In general, the separations are to have minimal pitching impulses, severance events that occur as rapidly as possible, a minimum amount of shrapnel or debris after separation, and a minimum of contamination of sensitive surfaces such as the solar cells or instrument sensors. Besides separating two parts, the mechanisms must provide primary load paths for all six degrees of freedom of the satellite pairs. Here a trade-off exists, since the kinematic criteria indicates the use of simple single point attachments, while the structural specifications require large section moduli for strength and stiffness.

6.2.1 SPIN-OFF SEPARATION MECHANISM

The principle difficulty in the spin-off separation is the need to control the time of separation very accurately. Spin-off separation timing implies simultaneity of the pairs release, and accuracy in the time lapse between the electrical signal from a sensor to the initiation ordnance and the completion of the mechanical separation. If the time lapse is either too long or too short, the spin-off velocity component along the orbit tangent will create excessive satellite array separation distances during the mission. The net requirement is then to perform the event very rapidly so that no significant errors are introduced.

To adequately control this timing problem in the spin-off maneuver, a component search of the available hardware was performed. In almost every case, the timing uncertainty for clamps or so-called separation bolts was in excess of the desired 1 millisecond uncertainty, at a 3σ confidence limit. The only pre-tested units which could provide this order of timing accuracy (and could produce data to confirm the performance) were exploding bolts employing a capacitive firing module. The essential properties of this system are the firing unit which has dual coaxial cabling running to the two explosive bolts. The current is discharged into exploding bridge wires which in turn initiated the ordnance and effected the bolt severance. While giving very rapid and accurate function times, these bolts tend to be less reliable than the slower acting variety. This is apparently due to the possibility of the wire exploding without igniting the detonator. Another dis-advantage of this approach is that its weight is about twice that of a comparable conventional explosive bolt.

A structural disadvantage exists for all of the systems employing a single bolt as the separation component, since the single bolt cannot provide high torsional rigidity when the satellite pair is connected. This is primarily because of the very large pitching moment of inertia of the satellite pairs. The torsional rigidity of the bolted mechanism is further compromised by the way in which the mechanisms hold the bolt. These mechanisms must be designed to ensure that the bolt is tightly clamped for all degrees of freedom of the satellite pair, requiring a complicated mechanism needing considerable adjustment and pre-stressing.

For these reasons, a torque tube was considered which is circumferentially severed by means of a flexible linear shape charge (FLSC). The main problem with this approach is that it is in the development stage and there is uncertainty in the prediction of characteristics of the separation such as severance time, pitching moments, and normal thrust. These parameters would have to be determined experimentally. Indications are that separation time uncertainties will be less than the desired 1 millisecond, but no directly applicable data exists.

The two separation drawings (Figure 37 and 38) show both the exploding bolt mechanism design (Figure 37) and the FLSC severed torque tube mechanism (Figure 38). The obvious trade-off considerations are the proven performance of the more complicated bolt assembly with its structurally elastic joint, compared to the very simple and structurally superior torque tube design employing a severance scheme which is currently a developmental item.

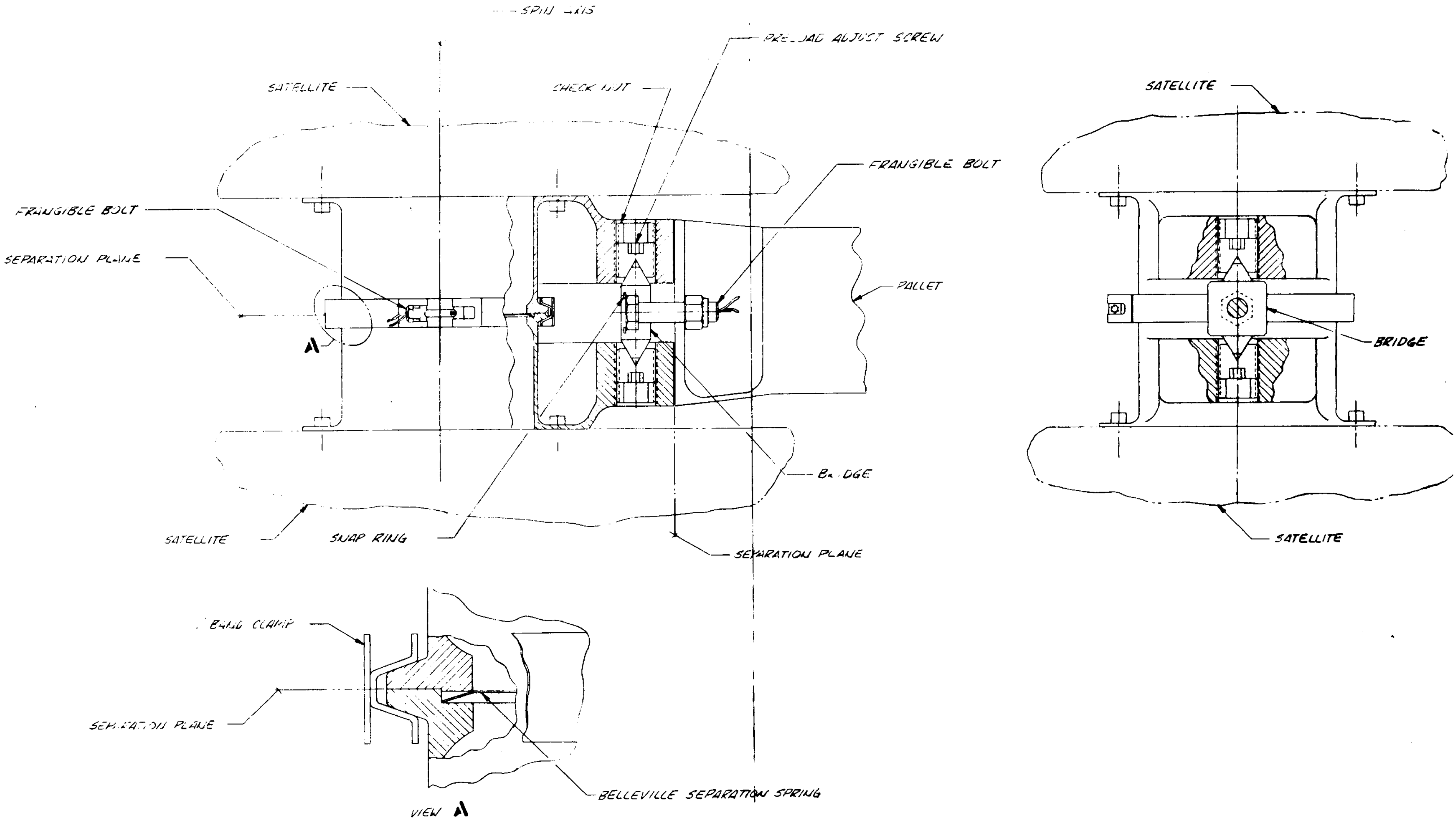


Figure 37. Explosive Bolt Separation Mechanism

Reduce
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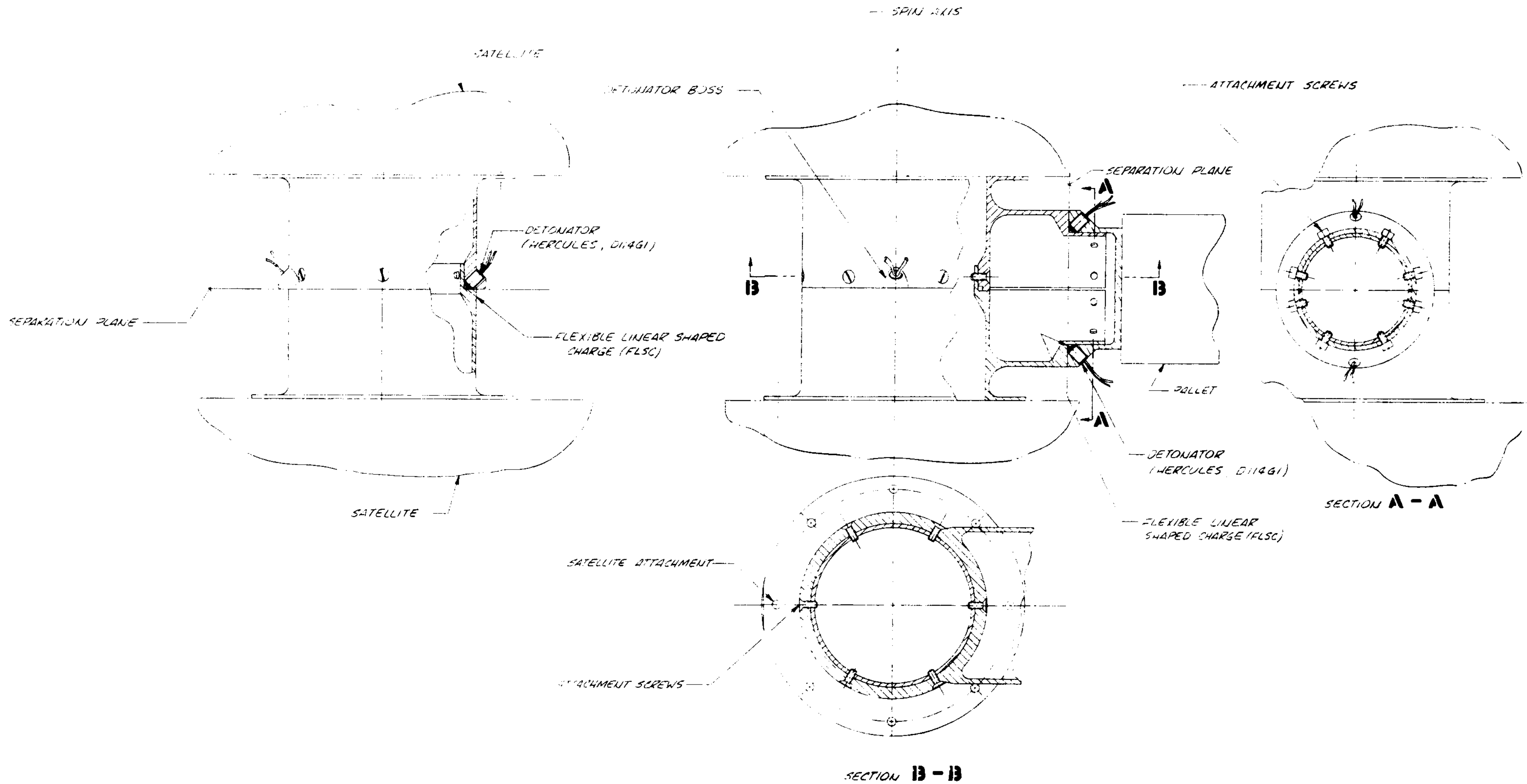

Figure 38. Shaped Charge Separation Mechanism

6.2.2 AXIAL SEPARATION MECHANISM

The axial separation of the satellite pairs into the individual satellites is a much simpler and more straightforward design problem. A purely mechanical design using a conventional Marmon clamp and a Belleville ring spring to give the slight axial impulse for separation are shown in one design. The advantages of this approach are similar to the bolt approach on the spin-off device; the system employs flight-proven techniques. It is however, somewhat complicated and will require considerable adjustment. The simplicity of the FLSC both in severing the connection tube and giving the desired small axial thrust is obvious.

6.3 BOOM DESIGN

The magnetic background of the magnetometer sensor is required to be quite small. A most effective way to satisfy this requirement is to provide a boom of sufficient length to displace the magnetometer sensor to a location of low satellite-induced magnetic background. This requirement has been met before with booms (e.g., IMP AND PIONEER VI), but not under the peculiar design constraints of the multiple satellite. These constraints are:

a. The packaged configuration for the booms must result inno intersatellite or pallet-satellite interference during separation;

b. The boom must be of minimum weight so that the satellite is not overly despun upon deployment.

With small, lightweight satellites such as these, a single boom can best satisfy the packaging and dynamic requirements. Although the use of a single boom on a spinning satellite is somewhat unusual, examination of the effects on spin stability characteristics indicates no serious adverse consequences. The deployment of a single boom will cause the satellite to depart from axial symmetry with respect to inertial properties, i.e., the moment of inertia about all lateral axes will not be equal but will vary with the quadrant of the axis. However, the basic criterion for stability with respect to energy dissipation effects remains unchanged; the satellite is stable when spinning about its major principal axis, i.e., the body axis about which the moment of inertia is greatest.

The satellite will be designed so that the major principal axis is aligned with the longitudinal reference axis prior to deployment. The boom will be deployed laterally at the station of the center of mass. Hence the deployment does not change the station of the satellite's center of mass, and since the contribution of the moved elements to the axial products of inertia, is zero before and after deployment, the direction of longitudinal principal axis remains aligned with the direction of the longitudinal reference axis. Since the deployment of the boom increases the moment of inertia about the longitudinal principal axis as much as it increases the moment of inertia about any lateral axis, the longitudinal principal axis remains the major principal axis. To the extent that the deployment of the boom may be slightly off-nominal, the major principal axis after deployment may be slightly tilted with respect to the longitudinal reference axis. It is expected that this tilt can be kept below 1 degree.

The deployment of the boom will displace the satellite's center of mass laterally in the direction of boom deployment. Current estimates indicate that this displacement will be about 4 cm; however, it will be accurately predictable, and can be anticipated in the location of any specific components, if necessary.

The departure from axial symmetry does result in a somewhat different dynamic behavior in the presence of a coning condition. Instead of remaining fairly constant, the cone angle oscillates between upper and lower bounds. The main significance of this difference is its effect on the design of a precession damper for the satellite. Because of the departure from inertial axial symmetry, the effectiveness of a traveling mass precession damper will be affected by the quadrant in which the damper is mounted. While it may not be possible to take advantage of the boom arm to enhance the effectiveness of the damper, no particular difficulty is anticipated on this account, since the requirements on the satellites' precession dampers are relatively modest.

The requirement for no change in station of the satellite COM due to boom deployment is directly satisfied if the boom is deployed in a plane perpendicular to the spin axis. This "yo-yo" deployment mode introduces no bending of the boom components until the boom is fully extended. This type of deployment motion has two phases. The first is a tangential growth until the boom is fully extended, during which time the components experience only tension. The second phase occurs when the final position is reached and the radial kinetic energy must be absorbed in flexural strain energy. The conventional axial deployment, where the booms are constrained to deploy in a plane containing the spin axis, constantly subjects the boom elements to transverse bending moments. Thus from a deployment survival approach, a radially deployed boom can be lighter.

6.3.1 HINGED LINK BOOM

A conventional design for the single boom deployment is the hinged link boom. This design is shown in Figure 39. To satisfy the deployment criteria, it is necessary to have the center of mass of the hinged boom always at the same station. This requires a minimum of two short links and one long link. (The design in Figure 39 does not have these characteristics). Since this, design can not be packaged, the next possibility is a 5-link boom having two short links and three longer links. This system is deployed in the same manner as the axial booms which will introduce the Coriolis bending moments during deployment. In general, this design is difficult to package and does not adapt well to spin deployment testing.

6.3.2 HINGED SEGMENT BOOM

A modification of the hinged link boom to an eight section multilink boom is shown in the second design concept. (Figure 40). Here the links are wrapped radially around the satellite. The deployment mode is a two-phase yo-yo motion. The tangential phase strings the links tangentially and locks them while under tension loading. The second phase then swings the extended links radially, still only under tension. In addition, the drawing shows a tension line which can be used to assist in deploying the boom if

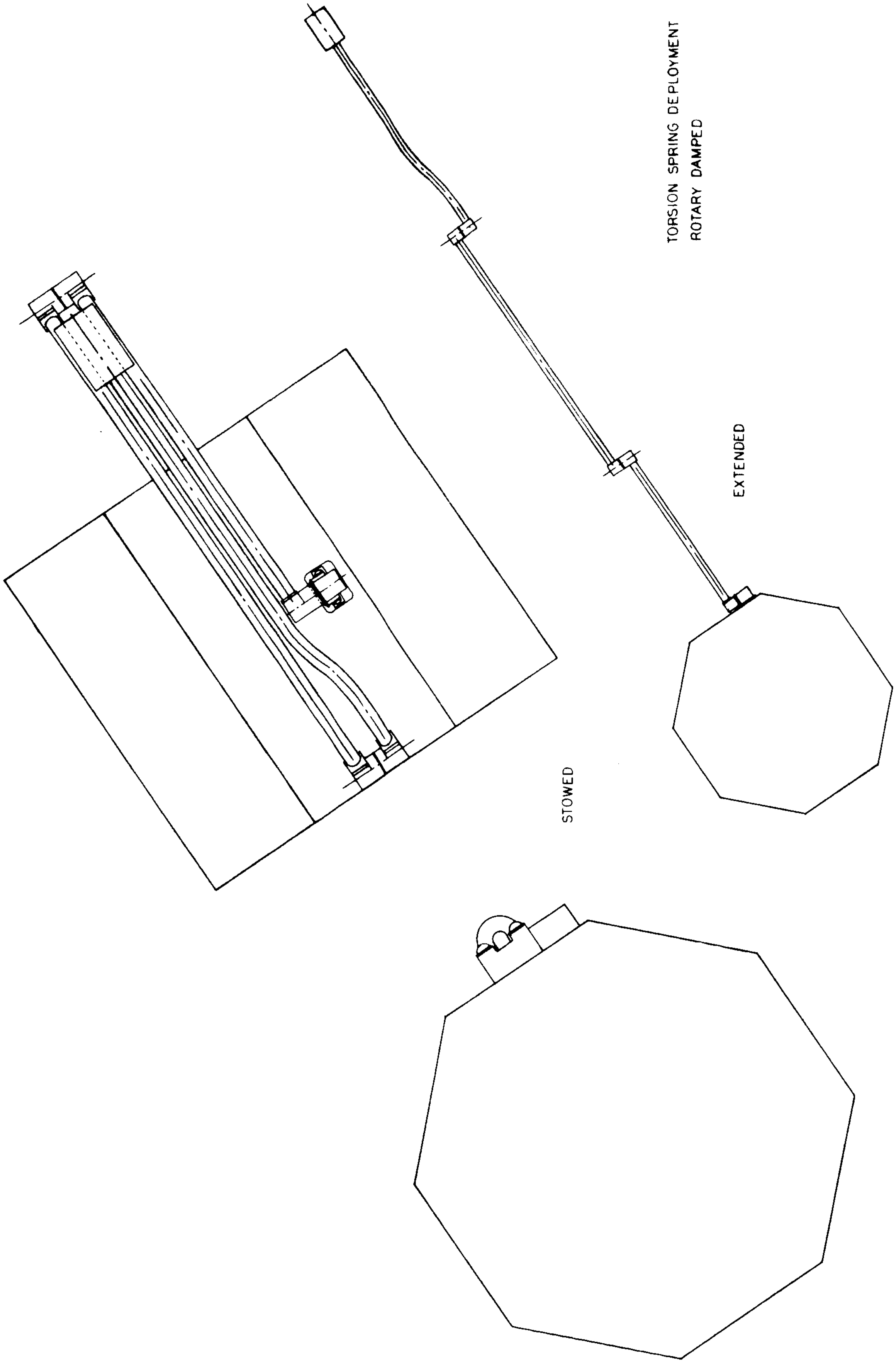

Figure 39. Hinged Link Boom

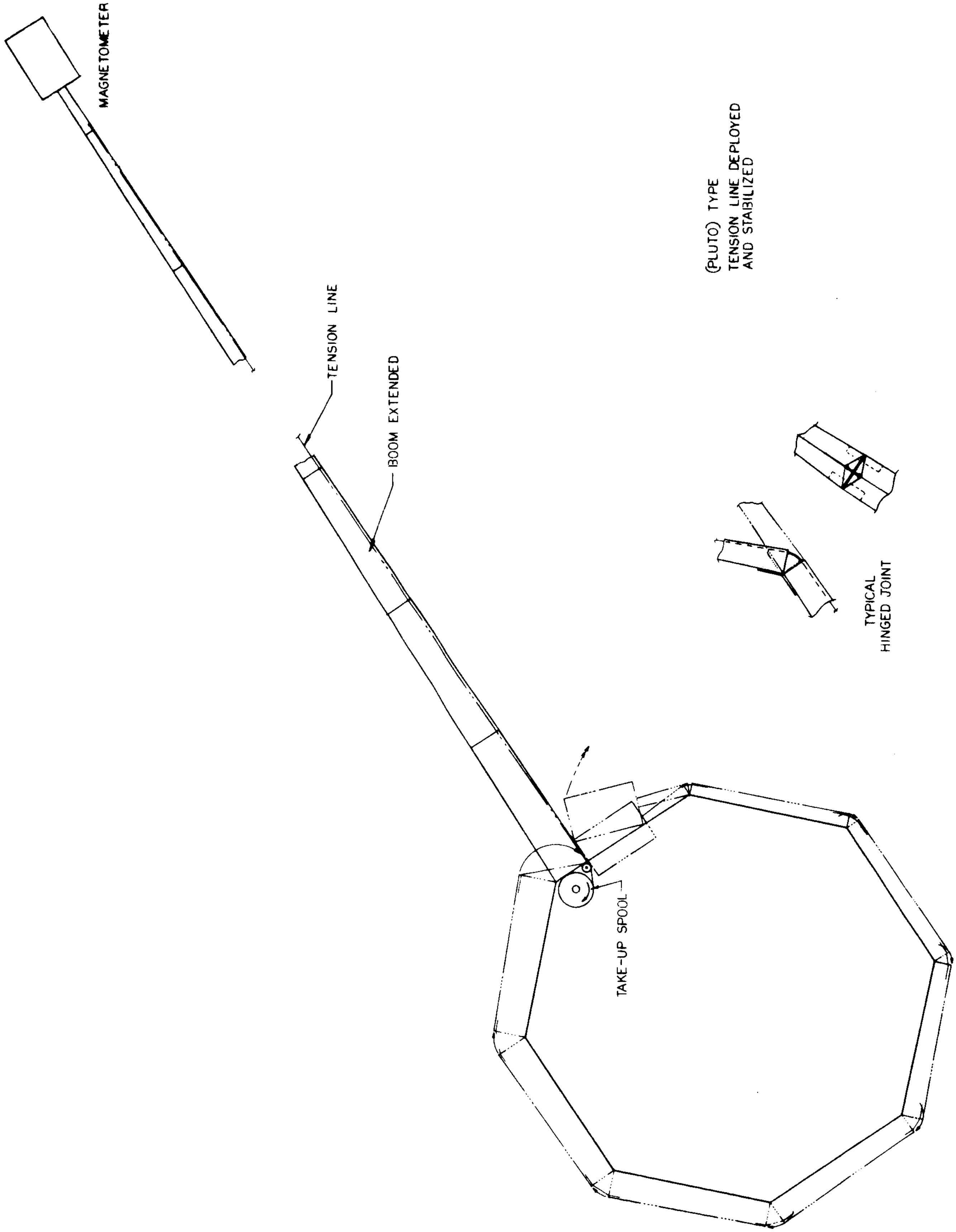

Figure 40. Hinged Segment Boom

the spin rate is too low. This tension can stabilize the boom when fully extended. Coiled springs can also be used to assist the deployment.

6.3.3 TETHERED WHIP BOOM

A third concept extrapolates the rigid multi-link design with coiled springs into a continuous elastic "fishpole" design which is wrapped around the satellite, as shown in Figure 41. This design must be made of non-metallic isoelastic materials, but will not require joints, hinges, or springs. This can be a distinct advantage in maintaining magnetic cleanliness. The primary difficulty with the "whip" design is the need for rigidity in the boom when fully deployed combined with sufficient flexibility to wrap around the two foot diameter satellite.

6.3.4 TELESCOPING BOOM

The final concept presented is the telescoping boom (Figure 42). This boom creates the least disturbance to the satellite and is completely controllable during the deployment. In addition, the absorption of energy at the completion of deployment is taken in tension instead of in bending strain energy as in the other designs. The inherent difficulty with this design is a weight penalty due to the need to overlap the telescoping segments and the need for an active deployment drive mechanism to overcome friction. Another potential problem is the difficulty in packaging before deployment. The design shown in Figure 42 utilizes a fixed length lead screw which is successively threaded from one segment to another as each segment is deployed. It is possible that the boom could be deployed by centrifugal acceleration if the friction between segments can be minimized. A pneumatically deployed boom of this type with two segments was flown on the early Ranger spacecraft.

6.4 THERMAL CONTROL

The multiple satellite configurations have been designed to utilize passive thermal control. A major problem in passive thermal control design is the selection of thermal control finishes that will maintain satellite

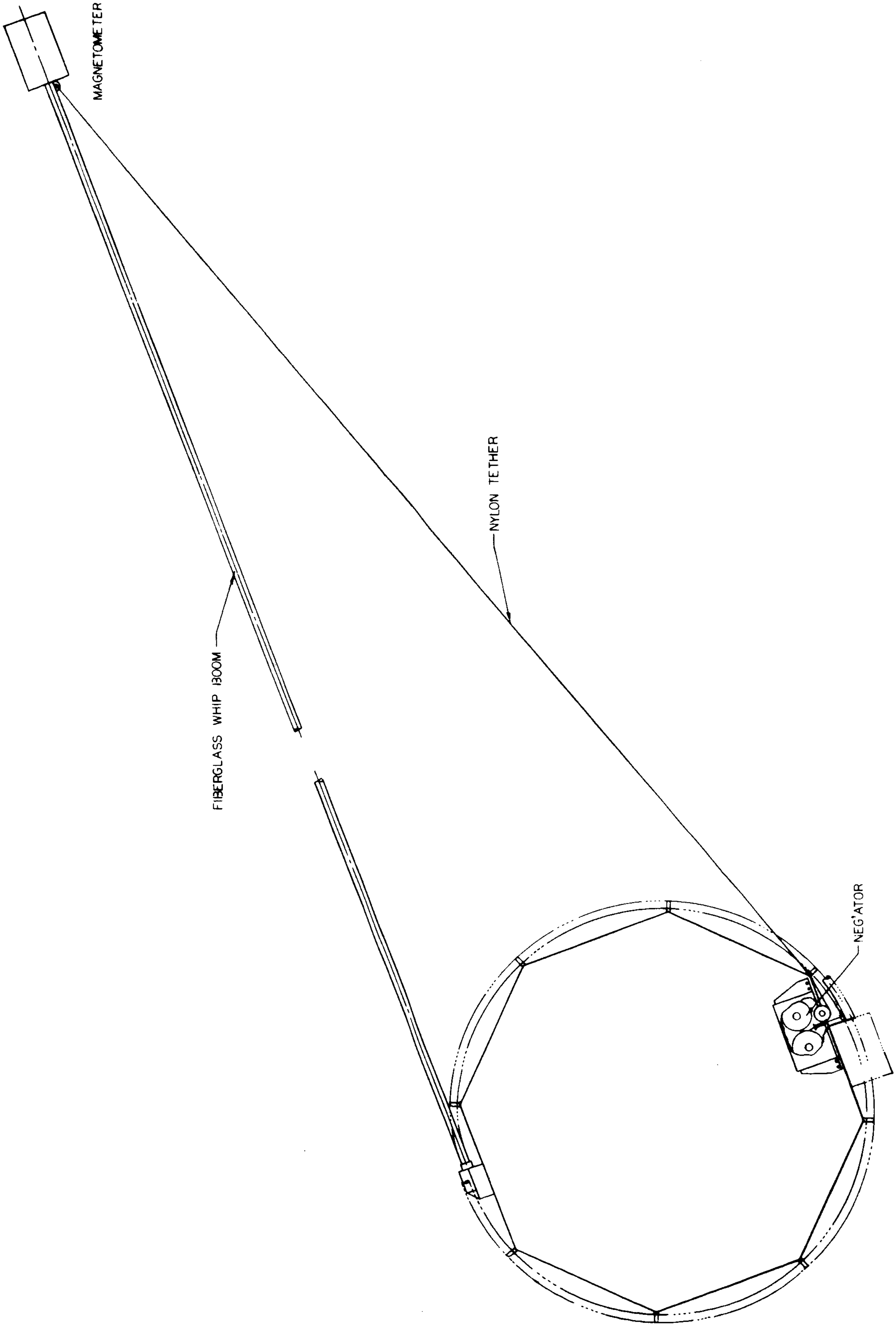

Figure 41. Tethered Whip

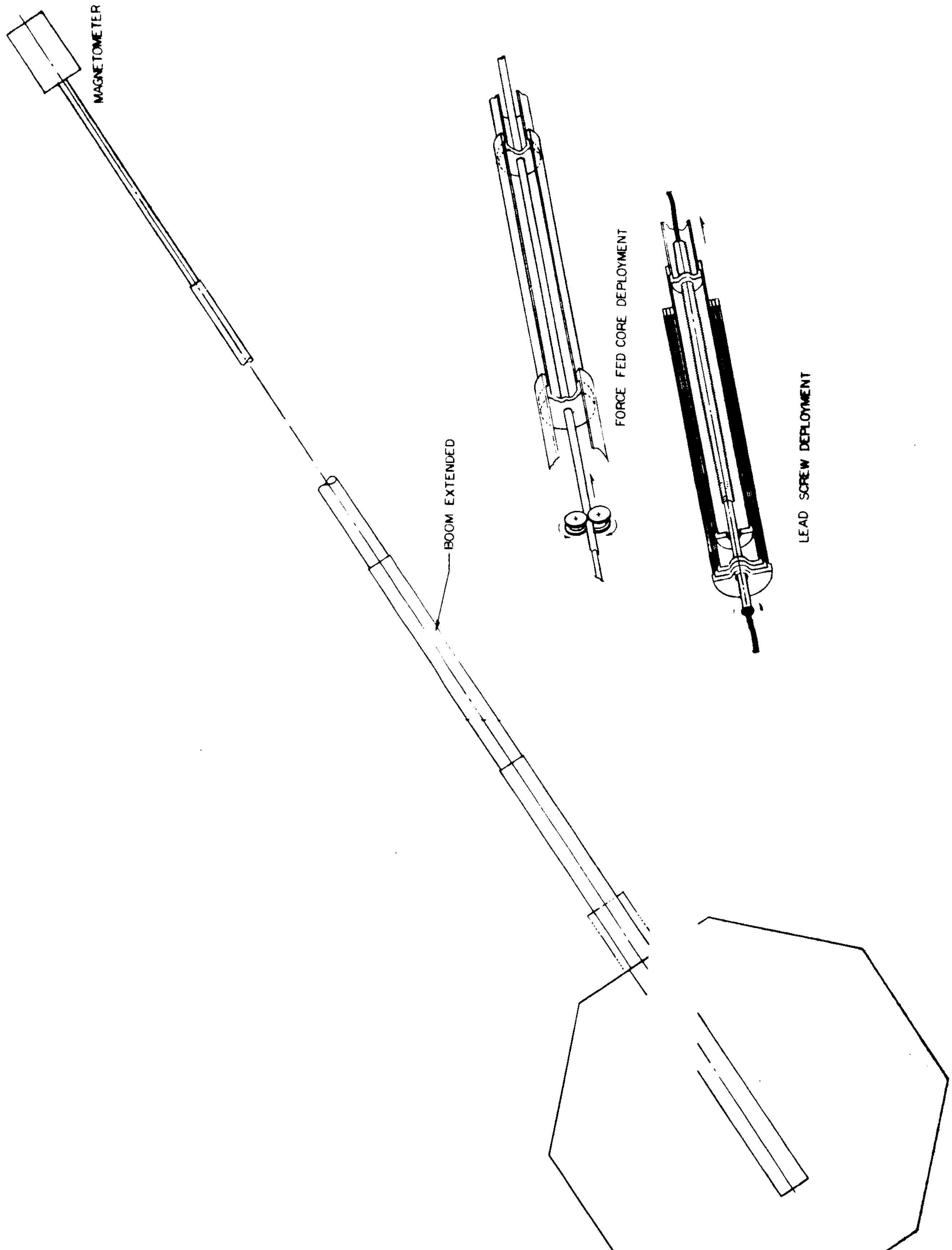

Figure 42. Telescoping Boom

temperatures within tolerable limits while the satellite receives maximum solar and planetary inputs and also while the satellite passes through the earth shadow. Satellite equilibrium temperatures were calculated throughout an orbit for the two orbital extremes of the sun at apogee and perigee, assuming a homogeneous satellite[29]. Various surface finish combinations were used in the analysis. Figure 43 shows the equilibrium temperatures for the two orbital cases, and the surface conditions assumed. The temperature variation is considered acceptable except for the case when the sun is at apogee with occultation. Because this is at the end of the mission and the orbit selected will not result in occultation as severe as that simulated, the poor thermal control indicated for this case is not considered detrimental to the mission. More detailed calculation assuming a nonhomogeneous satellite show that the temperature variation can be reduced through simple insulation design practices. As the design changes and becomes better definitized, more detailed thermal analyses will be required to verify that passive thermal control can be utilized. However, unless major design variations are made, it appears that passive thermal control is feasible for the multiple satellites.

6.5 POWER SYSTEM

Power requirements have been established including profiles for the pallet, for the satellites in the real time data mode; and for the satellites in the record mode.[30]

Pallet power requirements were determined to be 95 watt-hours which can be provided by silver-zinc batteries weighing about 2.25 lbs.

The satellite power requirement is established by the real time data mode. Figure 44 shows the real time data mode power profile. The average power required by the subsystems is 17.5 watts. The solar array is sized for 22 watts which provides the 17.5 watts, an addition 0.5 watts for battery recharge, 10% power conversion efficiency loss and 8% magnetometer boom shadowing power loss. The boom shadowing effect was

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[29] See Appendix XXIV
[30] See Appendix XXV

UNIFORM EQUILIBRIUM TEMPERATURE HISTORY FOR AN ORBITING SATELLITE

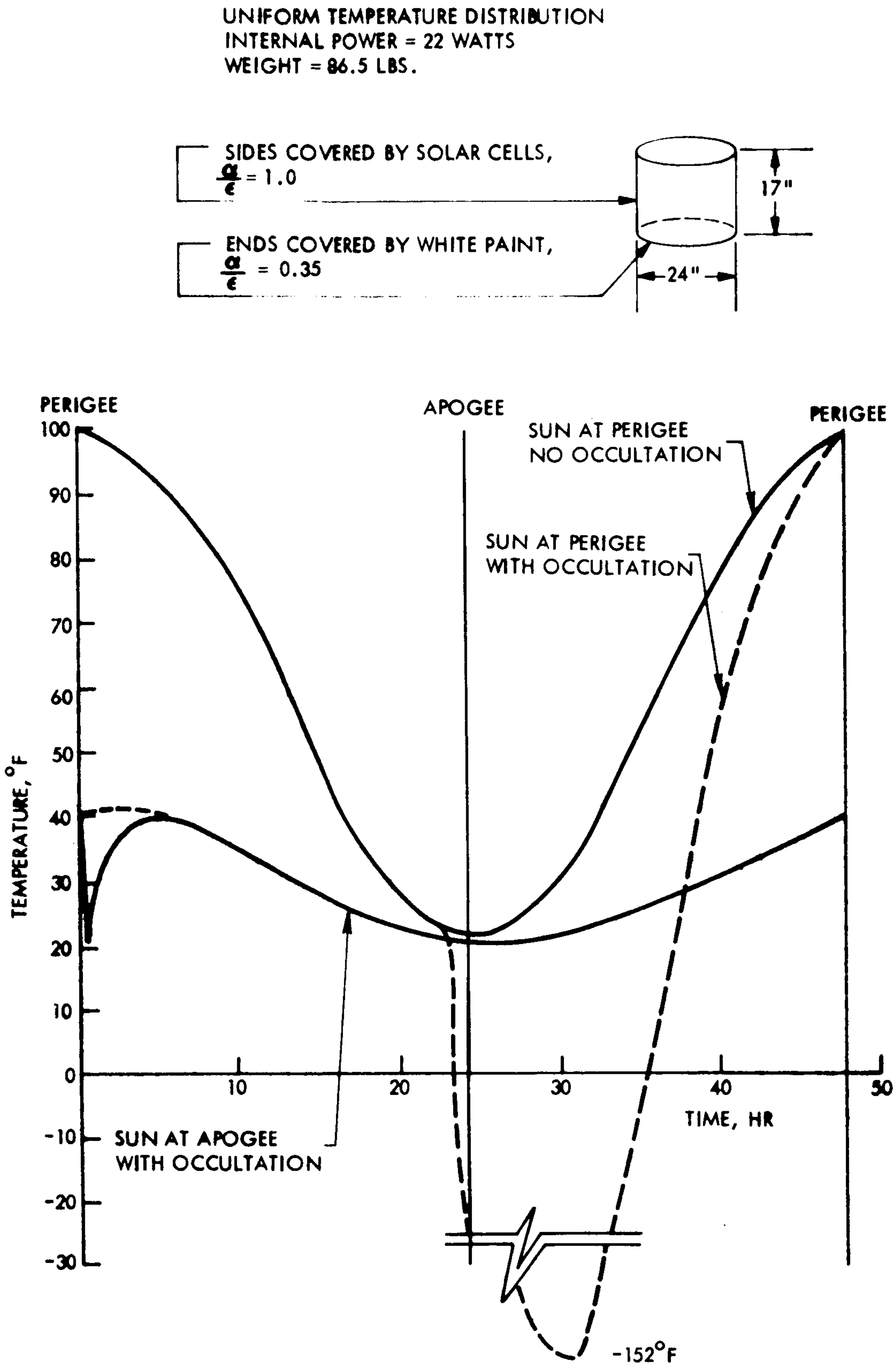
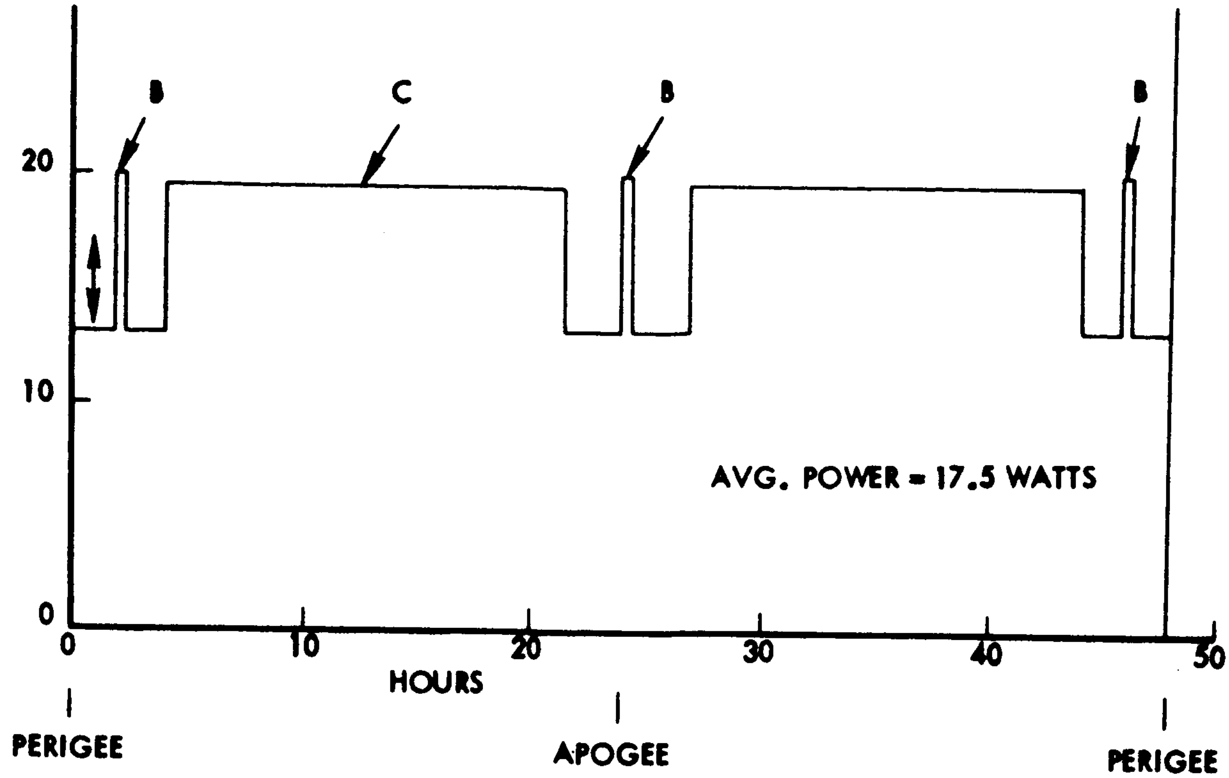


Figure 43. Temperature Profile



SUBSYSTEMS ON

A	B	C
INSTRUMENT COMMAND RECEIVER DATA HANDLING SYST. ASPECT SENSOR	SAME AS A + TRANSPONDER	SAME AS A + TRANSMITTER

Figure 44. Power Profile, Real-Time Data Mode

analyzed[21] the results are illustrated in Figure 45, where the dotted curve represents the shadowed solar array power output.

Figure 46 shows the record data mode power profile indicating lower power requirements than the real time data. Some real time transmission or additional record time could be added if desirable and if compatible with other subsystem capabilities.

The solar cell area to meet the 22 watt requirement has been calculated as 2.6 ft^2 with a weight, including structural backing, of 11.2 lbs.

6.6 COMPONENT AVAILABILITY

An objective of the Multiple Satellite Program requires maximum use of off-the shelf components. Use of such components would minimize development costs and enhance compliance with reliability goals through the benefits derived from prior experience. It is apparent, however, that each program has unique requirements and that modifications to existing equipment are usually necessary to meet the present program specifications. Such areas as solar panels, battery packs, separation ordnance, etc. require tailoring to specific program needs. Others, such as data processing and attitude control systems may be fabricated with minor modifications to existing equipment. A component list has been prepared which reflects the availability of major components; the list will be continually updated as the program progresses. A summary is given in Table 15 of the available hardware, previous programs which have utilized these components, and their manufacturers

6.7 DESIGN SUMMARY

The multiple satellite design effort has attempted to maximize the payload weight, volume and data rate allowable for the scientific instruments, while providing adequate support subsystems and meeting all other constraints. A summary of the system characteristics resulting from the design are the following:

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[21] See Appendix XXVI

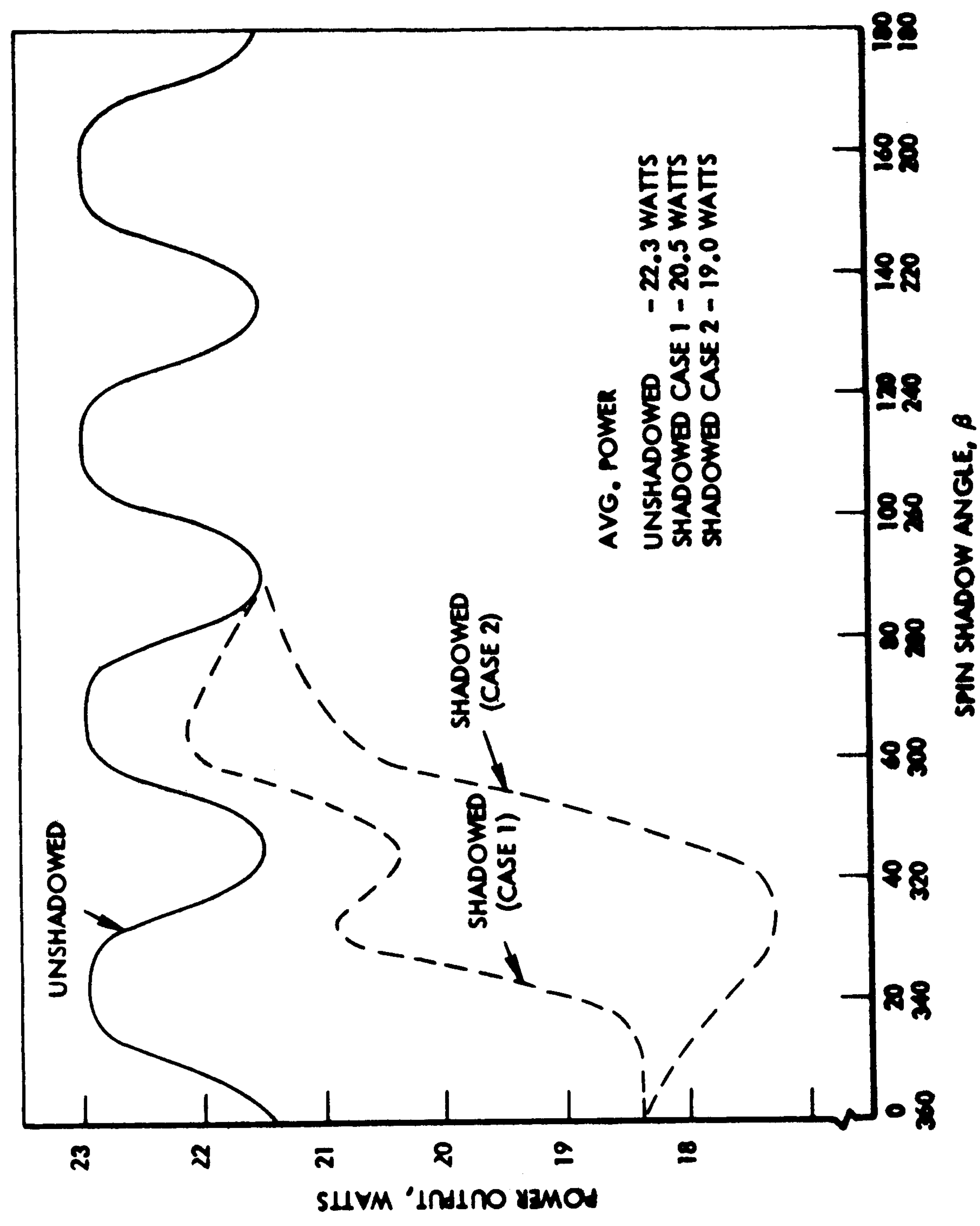
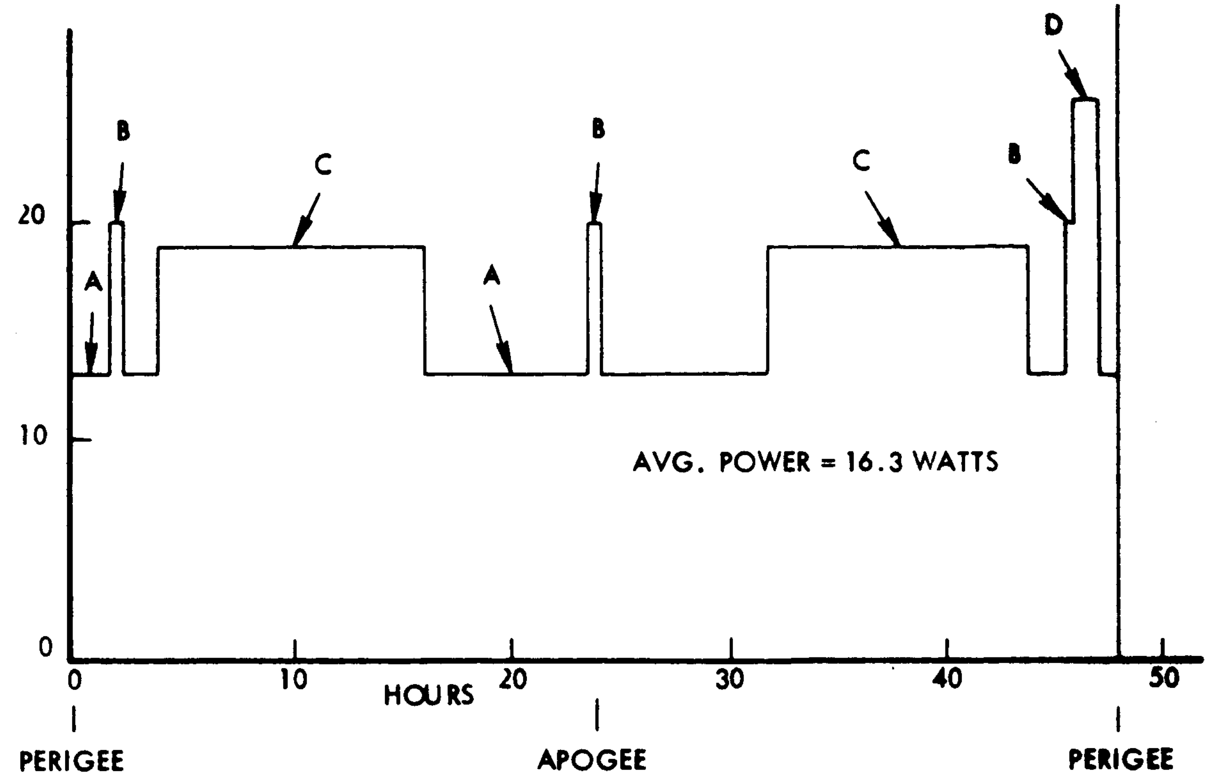


Figure 45. Maximum Power Loss Due to Boom Shadowing



SUBSYSTEMS ON

A	B	C	D
INSTRUMENTS COMMAND RECEIVER DATA HANDLING SYST. ASPECT SENSOR	SAME AS A + TRANSPONDER	SAME AS A + RECORDER	SAME AS A + TRANSMITTER AND RECORDER

Figure 46. Power Profile, Data Record Mode

Table 15

AVAILABLE COMPONENT SURVEY

Subsystem	Program	Vendor
ORIENTATION		
Sun Sensor	Pioneer, OV-3	Ball Bros, Bendix-Eclipse
Aspect Sensor		
Magnetometer	OV-3, OSO	Schonstedt, Marshall Labs
IR Sensor	TIROS, OGO	Barnes, ATL
Pneumatics	Syncom, AOSO, Pioneer	Kidde, Honeywell, TRW
COMMUNICATIONS		
"S" Band Transmitter		Conic, Vector, TRW
Command Receiver/ Decoder	OGO, Pioneer, OV-3	Conic, TRW, RS Electronic, AVCO
PCM System	Owl, Apollo, Ranger	SGC, Teledyne, Radiation
Tape Recorder	OV-3, UK Satellites, Gemini	Raymond, Kinelogic
Tracking Beacon	OSO, OAO, NIMBUS	Motorola, GSFC
POWER		
Solar Panels	OV-3, OAO, Pioneer	Spectrolab, Helioteke, Hoffman
Battery Pack	OV-3, Explorer, OAO, Pioneer	Eagle-Picher, G.E., Yardney, Gulton
Power Conditioner	OV-3	SGC, Honeywell
MISCELLANEOUS		
Explosive Bolt	Surveyor	Hi-Gear Corp.
Flexible Linear Shaped Charge		Conax Corp.
Cable Cutter		McCormick-Selph

1. <u>Instrument Allowances per Satellite</u>

Weight	26 lb
Volume	4600 in^3
Power	8 watts
Bit Rate	1050 bits/sec

2. <u>Payload</u>

Weight	399.4 lb
Maximum Diameter	52.0 in. (shroud limitations)
Maximum Length	46.0 in. (shroud limitations)
Moment of Inertia Ratio	1.17

3. <u>Satellite</u>

Weight	85.6 lb
Maximum Diameter	22 in.
Maximum Length	17 in.
Total Subsystem Power Available	18 watts
Boom Length	88 in.
Final Spin Rate	60 rpm
Component Temperature Range	20^o to 110^oF
Moment of Inertia Ratio -	
a) Boom retracted	1.44
b) Boom deployed	1.32

Section 7

CONCLUSIONS

The analysis and design efforts performed under this program have shown the multiple satellite mission to be technically feasible. Alternate mission modes were examined with the selection of an active pallet (or bus) design concept. The pallet orients and deploys the four satellites into a non-coplanar array which meets the scientific experiment requirements.

The THORAD/Improved Delta/FW4 launch vehicle will be available and operational when required by the Multiple Satellite Program, and can launch the payload into the orbit required. A common orbit has been selected which allows coverage of the transition region and near-interplanetary space, and provides substantial coverage of the subsolar region.

Alternate deployment methods were investigated with the selection of one which provides a non-coplanar array with the required separation distances. A pallet control system has been defined which can perform the required re-orientation maneuver.

A communications and data handling system has been specified which maximizes the bit rate while using developed subsystems. STADAN ground station availability was investigated and found to be adequate provided that on-board satellite data storage is incorporated; there is adequate coverage of all four satellites for command or data reception by the ground stations during each orbit of the mission.

The design which evolved from the study maximized instrument weight, volume and data rate for the four satellites. These capabilities are, for each satellite:

- 26 lb instrument weight
- 4600 in^3 instrument volume
- 8 watts instrument power
- 1050 bits/sec instrument data acquisition rate.

These capabilities are more than adequate for the performance of the mission.

Some of the specific problem areas which require early work to further definitize the multiple satellite system include:

1. Investigation of an integrated instrument complement.

2. Optimization of satellite separation and array characteristics.

3. Design and demonstration of the magnetometer boom.

4. Design and demonstration of the spin separation mechanism.

The multiple satellite contract contained the New Technology Clause. No new technology "reportable items" were generated under the current multiple satellite contract.

Section 8

RECOMMENDATIONS

Based upon the currently established feasibility of the Multiple Satellite Program, initiation of a Program Definition Phase is recommended. Prior to initiation of the overall system preliminary design, additional technical preparation is also recommended in the following specific areas:

1. Instrument Inegration Study
2. Separation Optimization
3. Detailed Array Analysis
4. Reorientation and Aspect Sensing Optimization
5. Boom Design and Demonstration
6. Spin-Separation Design and Demonstration

8.1 INSTRUMENT INTEGRATION STUDY

The primary instruments which were originally specified for the program included the Pioneer magnetometer and the Pioneer plasma probe. The design of these instruments was optimized for the Pioneer mission, which had significantly different scientific objectives and system constraints from those which exist for the multiple satellite effort. In particular, the cycle period for the plasma probe is considered to be too long for effective resolution of high speed disturbance propagation, and the on-board data processing for both instruments is designed to minimize the downlink rates over the extremely long Pioneer communication distances.

The suggested program would involve extensive coordination with personnel of the ARC Space Sciences Division. It will establish definitive requirements on the instruments as necessary to meet the multiple satellite scientific objectives, and will define the hardware and operational modifications to the existing instruments which are necessary for application to this program. Functional and interface characteristics of the on-board data processing for each existing and/or proposed instrument will be reviewed relative to the multiple satellite communication parameters. This effort will define the most effective integrated data subsystem for the Multiple Satellite Program

8.2 SEPARATION OPTIMIZATION

The multiple satellite mode of operation requires that precise separation of the satellites from the pallet be implemented in order to achieve the orbital array required.

Because of the limited scope of the current phase of the Multiple Satellite Program, it was not possible to analyze the satellite deployment and separation in great detail. A deployment mode was established which provided an acceptable satellite array, but sufficient analysis has not been performed to optimize the separation/array problem. Further deployment analysis coupled with array simulation is required to optimize the separation system, considering error and perturbation effects. The objective of the Separation Optimization Program is thus to analyze and optimize the separation mode considering satellite separation and resultant array characteristics.

8.3 DETAILED ARRAY ANALYSIS

The relative motion of the satellites after deployment for a specific deployment approach was analyzed during this program. Although an acceptable array was achieved, error and perturbation effects were not fully anticipated. In order to optimize the deployment, perigee altitude, and other array-related characteristics; further analyses of the array under varying conditions of deployment, orbit parameters, separation errors, initial conditions with respect to the sun position, perturbation effects, etc., is recommended. The purpose of this recommended program would be to study in more detail the satellite array. Maximum use would be made of computer programs to provide detailed time-varying relative effects between the satellites of the array.

8.4 REORIENTATION AND ASPECT SENSING OPTIMIZATION

The system design requires that the spinning spacecraft be reoriented from the attitude inherited by the boost vehicle to an attitude normal to the orbit plane. This maneuver is very similar to that performed by the Pioneer VI spacecraft, and has been analyzed in previous Space-General in-house studies. The feasibility of such a maneuver is not in question, and thus only preliminary work was performed under the current study to establish the required control and sensing systems. Considerably more work is required to optimize the reorientation system. Requirements should be established in further detail, considering the effects of orientation accuracy and procedures on deployment, array errors, and communications. Alternate modes of operation together with varying attitude control, precession damper, and aspect sensing designs must be considered. The proposed study effort would examine all of the above mentioned considerations integrally to optimize the complete reorientation/aspect sensing system.

8.5 BOOM DESIGN AND DEMONSTRATION

On satellites carrying magnetometer instruments for study of low level magnetic fields, it is necessary to extend the magnetometer sensor on a boom of significant length to minimize the background effects of the satellite itself. These booms have typically consisted of conventional multiple hinged arm segments. The weight vs length characteristics plus the deployment reliability for such multiple hinged booms have left much to be desired. A weight characteristic typically limits the boom length to some compromised value, both due to its effect on over-all satellite weight and also due to the large spin down ratios and moment of inertia ratio variations which result when a relatively heavy boom is deployed to a large radius.

Several unconventional approaches for magnetometer boom design have been identified in the course of the current program. These include two versions of a wraparound boom, one flexible and one with many short segments, plus one positive-extension telescoping type boom. It is suggested that further design work be completed for several unconventional boom concepts, and that the most promising approach be demonstrated by fabrication and test.

8.6 SPIN-SEPARATION DESIGN AND DEMONSTRATION

The system design concept which has been selected for emphasis in the Multiple Satellite Program includes a pallet subsystem which reorients the payload prior to satellite separation. The satellites are then spin-separated from the pallet in a maneuver which achieves the desired in-plane normal satellite separation component. The accuracy and reliability of this spin-separation maneuver is a key area relative to the over-all feasibility of the pallet system concept.

The spin-separation mechanism must provide the following functional capabilities:

a. Provide the structural support between the satellites making up a single pair and between the pair and the pallet.

b. Allow alignment and adjustment of the individual satellites to achieve the required static and dynamic balance for the satellite pairs, the pallet plus four satellites, and the pallet plus four satellites mounted on the third stage.

c. Provide for separation of the satellite pairs from the pallet in a highly accurate azimuthal direction, without disturbing the spin-axis attitude of the paired satellites.

d. The structural elements holding the two satellites in a pair must provide for initial spring separation, and for subsequent solid rocket velocity gain, all without disturbing the spin-axis attitude for the individual satellites.

A demonstration program for the selected separation mechanism is recommended to verify the accuracy of spin-off timing, separation velocity, and separation direction, as well as verification of attitude stability.